A PRELIMINARY DESIGN OF A STANDARDIZED SPACECRAFT BUS FOR SMALL TACTICAL SATELLITES

THESIS

Written by GSO & GSE team

AFIT/GSE/GSO/ENY/96D-1

DEPARTMENT OF THE AIR FORCE

AIR UNIVERSITY

AIR FORCE INSTITUTE OF TECHNOLOGY

Wright-Patterson Air Force Base, Ohio

A PRELIMINARY DESIGN OF A
STANDARDIZED SPACECRAFT BUS FOR
SMALL TACTICAL SATELLITES

THESIS

Written by GSO & GSE team

AFIT/GSE/GSO/ENY/96D-1

Disclaimer

The views expressed in this thesis are those of the authors and do not reflect the official policy or position of the Department of Defense or the U.S. Government.

AFIT/GSE/GSO/ENY/96D-1

A PRELIMINARY DESIGN OF A STANDARDIZED SPACECRAFT BUS FOR SMALL TACTICAL SATELLITES

THESIS

Presented to the Faculty of the School of Engineering

of the Air Force Institute of Technology

Air University

in Partial Fulfillment of the Requirements for the

Degree of Master of Science

Gerald Ashby, B.S.
Capt, USAF

Darren Buck, B.S.
Capt, USAF

Robert Carneal IV, B.S.
1Lt, USAF

Tansel Cokuysal, B.S.
1Lt, Turkish AF

Ahmet Donmez, B.S.
1Lt, Turkish AF

James From, B.S., M.A.
Capt, USAF

Todd Krueger, B.S.
Capt, USAF

Brian Robinson, B.S.
1Lt, USAF

November 1996

Approved for public release; distribution unlimited

Preface

The following document is the culmination of the work performed by a team of eight graduate engineering students assigned to the Air Force Institute of Technology (AFIT). The students compiled this document while performing a systems engineering design study to create a small standardized tactical satellite bus for the Phillips Laboratory. This document is divided into three separate volumes. Each volume is an integrated element of the student thesis but it can also serve as a stand alone document.

The first volume is the Executive Summary. The purpose of the Executive Summary is to present a synopsis of the design study results to the sponsor at the Phillips Laboratory. This volume includes information on the methods employed during the study, the scope of the problem, the value system used to evaluate alternatives, tradeoff studies performed, modeling tools utilized to create and analyze design alternatives, recommendations and implications of the alternatives, and areas where future research should be considered.

The second volume is a detailed account of the design process. The steps of the team's innovative design process and the team organization are initially presented. Each phase of the design study is discussed in subsequent sections. Phase I provides accounts of the team's initial attempt to apply a well known systematic approach to satellite design. Efforts concentrate on defining the problem posed by the sponsor. "First cuts" at developing analysis tools and models are performed. Additionally, different alternatives are generated as possible solutions to the problem. An initial analysis and evaluation is performed to define an initial solution space, and to verify the analysis tool. Phase II is an iterative step in the design process and serves as a reservoir for the team's most meaningful work. The team realized that a new systematic approach had to be applied to the study. This phase provides the results of the application of that innovative approach. It is here that the understanding of the problem is further refined and decisions are made that limit the scope of the study. The objective hierarchy is further developed and a value system is created as a method for measuring each design alternative. Information is collected on satellite designs and satellite subsystems. Tradeoffs are performed to determined the best methods and components to be used in the alternatives. A model is created and design alternatives are generated. System analysis is performed on the alternatives using the value hierarchy, and results are generated. Sensitivity analysis is performed on the alternatives, and implementation recommendations are provided to the sponsor.

The third volume provides details on the tools developed to build a satellite and to analyze the design. There are three sections to this volume. The first section describes the model's philosophy and presents details on the purpose and operation of each module of the model. Mathematical formulae and module architecture are also described in this section. The second section is a user's guide to operating the model. Specific details of the sequence to be used and information required to run the model are provided in this discussion. The final section of this volume is the actual code of the model. The code is contained in an annex and is maintained by AFIT's Aeronautics Department at Wright-Patterson AFB, Ohio. The code can be provided to allow future modelers to understand and refine the work that has been accomplished.

Acknowledgments

The systems engineering design team would like to thank all the people who have provided their guidance, support, instruction, and personal time to ensure that the design study was a success. Special thanks go to the team's advisors, Lt Col Stuart Kramer, Maj Ed Pohl, and Dr Chris Hall. The team also realizes that it took more than just the team members to make the study meaningful and complete. Therefore, we wish to acknowledge the efforts of Lt Col Stan Correia, Maj Brad Prescott, Maj Scott Thomason, Doug Holker, Edward Salem, Lt Mike Rice, Capt Joel Hagan, Dave Everett, Col (ret) Edward Nicastri, Lt Col Brandy Johnson, Richard Warner, Linda A. Karanian, and the Space Warfare Center for the assistance and expertise they provided throughout our study. Finally, we would like to thank our families, who supported this effort with their patience and understanding: Sheri Carneal; Sedef and Sena Cokuysal; Rebecca, Austin and Travis From; Donna Krueger; and Coleen Robinson.

The Systems Engineering Team

Satellite Skunkworks

VOLUME II: DESIGN PROCESS

LIST OF FIGURES

LIST OF TABLES

Symbols

a	The semi-major axis (km)
Asa	Solar area (m2)
Asc	Surface area of component
CG	Center of gravity
C_r	Capacity required
d	Density (gr/cm^3)
D	Degradation per year
Fet	Configuration factor
g	Gravitational constant (9.806 m/sec^2)
GaAr	Gallium arsenide
H	Angular momentum
Het	Earth's emitted IR
Hsu	Solar constant
I_d	The inherent degradation
I_{sp}	Specific impulse (m/sec)
I_t	Total impulse (m/sec)
I_original	Original inertial matrix for a given sphere, cylinder, cone, etc.
I_total	Overall satellite inertial matrix
k	Specific heat ratio
LC^3	Linear, charge-current-control
L_d	Lifetime degradation
m_o	Initial vehicle mass (kg)
m_p	Mass of propellant consumed (kg)
M_{sa}	Mass of solar array
O/F	Mixture ratio for bi-propellants (m_{ox} / m_{fuel})
P	Pressure (Pa)
P_0	Power output (W/m^2)
PAS	Projected solar area
Qds	Incident solar energy on the satellite (α*PAS*Hsu)
Qer	Reflected solar energy (sun-earth-satellite)
Qet	Earth emitted radiation (ε*Fet*Asc*Het)
Qint	Heat output of the subcomponent
r	Radius
r_x	The transformation matrix converting the object's local frame of reference to the satellite's launch cg frame of reference

R	Gas constant (J/(Kg.K))
R_b	Blow-down ratio (V_{gf} / V_{gi})
R_m	Mass ratio (m_o / m_f)
SA	Solar array
Si	Silicon
SL	Satellite design life
SR	Shunt regulator
t	Thickness
T	Temperature (K)
Tsc	Temperature of subcomponent
TP	The orbital period (min)
TP_d	The minimum daylight period (min)
TP_e	The maximum eclipse period (min)
X_d	The efficiency of the path directly from the arrays to the loads.
X_e	The efficiency of the paths from the solar arrays through the batteries to the individual loads.
α	Solar absorptivity of the subcomponent
DI	Required specific impulse during mission of satellite (m/sec)
DV	Required delta velocity during mission of satellite (m/sec)
ε	IR emittance of the subcomponent
ϕ	The eclipse rotation angle(deg)
μ	The gravitational parameter (3.986012 x 105)
ρ	Angular radius of the earth (deg)
σ	Stefan-Boltzman constant (5.67x10-8 W-m-2-K-4)
s	Allowable stress

Abbreviations

AF	Air Force
AFIT	Air Force Institute of Technology
AFSCN	Air Force Satellite Control Network
ADCS	Attitude Determination and Control System
ARPA	Advanced Research Project Agency
ASAT	Anti satellite
ATSSB	Advanced Technology Standardized Satellite Bus
BER	Bit error rate
BOL	Begin of life
BW	Band width
CAD	Computer aided design
CCD	Charge coupled device
C&DH	Command and data handling
CDM	Chief decision maker
CEO	Chief executive officer
CTO	Chief technical officer
CER	Cost estimation relationship
cmds	commands
CMG	Control-moment gyro
COTS	Commercial off the shelf
CTA/STEP	Space Test Experiment Program
dB	Decibel
DET	Direct energy transfer
DOD	Depth-of-discharge
EO	Electro-optical
EOL	End of life
EM	Electro-magnetic
FMC	Financial Management and Comptroller
EPDS	Electrical Power and Distribution Subsystem
FY96$M	Fiscal year 1996 million dollars
GERM	General-Error Regression Model
GUI	Graphic User Interface
HAPS	Hydrazine Auxiliary Propulsion System
HETE	High Energy Transit Experiment
HTML	Hyper Text Markup Language
IFOV	Instantaneous field of view

IMU	Inertial measurement unit
IR	Infrared
IRQ	Interrupt request
KISS	Keep it short and simple
LASER	Light Amplification by Stimulated Emission of Radiation
LEO	Low Earth orbit
LEV	Launch Equipment Van
LIDAR	Light Detection and Ranging
LMLV	Lockheed Martin Launch Vehicle
LOWTAC	Low Tactical
LSE	Launch Support Equipment
LV	Launch vehicle
LWIR	Long wave infrared
m	Mass
MAXTAC	Maximum Tactical
MB	Megabyte
MIDTAC	Medium Tactical
MOE	Measure Of Effectiveness
MML	Mean mission life
MPE	Minimum percentage error
MSI	Multispectral-imaging
MSTI	Miniature Sensor Technology Integration
MTBF	Mean Time Between Failure
MUPE	Minimum unbiased percentage error
NASA	National Aeronautics and Space Administration
NIR	Near infrared
NiCd	Nickel - Cadmium
OLS	Ordinary least squares
OSC	Orbital Science Corporation
PPT	Peak-power tracking
PCSOAP	Personal Computer Satellite Orbital Analysis Program
RAM	Random access memory
RDT&E	Research, development, test and engineering
RER	Requirements estimation relationships
SAR	Synthetic aperture radar
SOH	Spacecraft state-of-health
SGLS	Space to ground link
SMAD	Space Mission Analysis and Design
SMC	Space and Missile System Center
SNR	Signal to noise ratio

SOS	Satellite operating system
SOW	Statement of work
SPIG	Spacecraft-To-Payload Interface Guideline
SSCM	Small Satellite Cost Model
SSLV	Single stage launch vehicle
SSRM	Single stage rocket motor
TFU	Theoretical first unit
TSS/SPI	Tactical Support Satellite/Standard Payload Interface
TT&C	Telemetry, Tracking and Commanding
TWTA	Traveling Wave Tube Amplifiers
U.S.	United States
USCM	Unmanned Space Vehicle Cost Model
UV	Ultraviolet
VIS	Visible
V	Volume
VSD	Value System Design

AFIT/GSE/GSO/ENY/96D-1

Abstract

A PRELIMINARY DESIGN OF A STANDARDIZED SPACECRAFT BUS FOR SMALL TACTICAL SATELLITES

Current satellite design philosophies concentrate on optimizing and tailoring a particular satellite bus to a specific payload or mission. Today's satellites take a long time to build, checkout, and launch. Space Operations planners, concerned with the unpredictable nature of the global demands placed upon space systems, desire responsive satellite systems that are multi-mission capable, easily and inexpensively produced, smoothly integrated, and rapidly launched. This emphasis shifts the design paradigm to one that focuses on access to space, enabling tactical deployment on demand and the capability to put current payload technology into orbit, versus several years by today's standards, by which time the technology is already obsolete. This design study applied systems engineering methods to create a satellite bus architecture that can accommodate a range of remote sensing mission modules. System-level and subsystem-level tradeoffs provided standard components and satellite structures, and an iterative design approach provided candidate designs constructed with those components. A cost and reliability trade study provided initial estimates for satellite performance. Modeling and analysis based upon the Sponsor's objectives converged the designs to an optimum solution. Optimum design characteristics include a single-string architecture, modular solar arrays, an internet-style command and data handling system, on-board propulsion, and a cage structure with a removable frame for easy access to subsystem components. Major products of this study include not only a preliminary satellite design to meet the sponsor's needs, but also a software modeling and analysis tool for satellite design, integration, and test. Finally, the report provides an initial implementation scheme and concept for operations for the tactical support of this satellite system.

Authors: Capt Gerald F. Ashby
Capt Darren J. Buck
1st Lt Robert W. Carneal IV
1st Lt Tansel Cokuysal
1st Lt A. Tuna Donmez
Capt James A. From
Capt Todd C. Krueger
1Lt Brian I. Robinson

Committee: Lt Col Stuart Kramer, Maj Ed Pohl, Dr Chris Hall

Sponsor: Lt Col James Rooney, PL/WS

1. Report Overview

This document provides the results of a group design study performed at the Air Force Institute of Technology. The team of eight graduate engineering students examined the design of a generic, standardized spacecraft bus for small tactical satellites. The project was sponsored by LtCol James Rooney of the United States Air Force's Phillips Laboratory in Albuquerque, New Mexico. Similar design studies have been completed by various companies and laboratories, but to date success has been limited. Phillips Laboratory's goal was to seek a "clean-sheet" approach to the design of a cost-effective satellite bus. Several design characteristics were suggested by the sponsor and were considered throughout the project. These characteristics included modularity, flexibility, robustness, and operability. These characteristics have been treated as guidance in developing objectives and alternative design architectures and were not treated as hard requirements.

This is the second volume of a three volume report. Volume I is an Executive Summary of the work performed by the design team. Volume II provides greater detail of the work and includes the theory and analysis behind the team's approach to the problem. Volume III is an in-depth explanation of the modeling performed for the project.

This volume is divided into three sections. The first section discusses the details of the team's design process that was created to approach the problem. The second section of this document, Phase I, records the team's first application of the a systematic approach to the design study. Details are provided on the initial methodology applied to the problem and preliminary satellite design information is presented. The work performed in this section of the report established a baseline for further iterations of the design approach and guided the team to new areas of research.

Phase II, the third section, is a repository for the majority of the work performed by the design team. This section documents the work performed in the second iteration of the systematic approach. During this portion of the study, the scope of the problem is refined and enhancements are made to the steps of the design process. Different design alternatives are presented and analyzed using the objective hierarchy developed as part of the process. The results of the analysis, and subsequent sensitivity analysis, are also provided. The section ends with a discussion of decision making and the implementation of the results.

2. Design Process

The design team recognized the need for a well-defined, iterative, systematic design process to approach the problem logically. The design team was familiar with two well-known systematic approaches, Hall's seven-step process to systems engineering (Hall, 1969:156) and the space mission design approach described in the Space Mission Analysis and Design (SMAD) textbook (Wertz and Larson, 1992:1).

Hall's systematic process has been a standard systematic approach for almost four decades. This process is well understood and can be applied to many different engineering problems. The Hall method is an iterative seven-step process (refer to Figure 2-1). These steps are: problem definition, value system design, system synthesis, system analysis, optimization, decision-making and implementation (Hall, 1969:157).

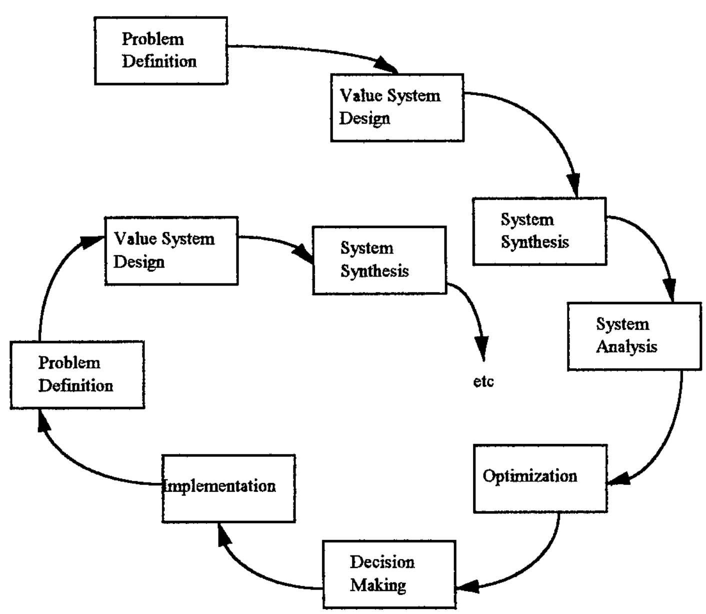


Figure 2-1: Hall's Seven-step Approach

Each step of Hall's approach is influenced by the actions taken in the other steps. The process' iterative nature forces refinement in each step as the process continues. Hall's fundamental framework follows a logical sequence that allows the user to define and constrain the problem, create an evaluation tool using the decision-maker's values, and generate possible solution alternatives. The framework also permits the user to create models and perform simulations as a means of quantifying aspects for each alternative. The quantified values serve as an input into the evaluation tool. Once the basic modeling is accomplished, different aspects of each possible solution are further refined in an attempt to optimize each alternative. Hall's process also allows the user to perform sensitivity analysis on each of the alternatives before the decision-maker is presented with the results of the system evaluation. In the decision-making step, the decision-maker applies his subjective values and risk preferences to select an alternative. With an alternative selected, a plan for implementation is created. The Hall process is complete once an adequate implementation strategy is accepted by the decision-maker.

The SMAD approach is well-known to contemporary satellite designers (Warner, 1996). The SMAD text and the process it describes is a compilation of the first thirty years of satellite design experience. In general terms, the SMAD process can be considered the classic approach to satellite design because the approach is based on the premise that the satellite's mission drives the design of the satellite bus. The SMAD approach is iterative and consists of four broad areas. These broad areas are 1) define objectives, 2) characterize the mission, 3) evaluate the mission, and 4) define requirements (Wertz and Larson, 1992:2).

Table 2-1: Space Mission Analysis and Design Process

Step	Sub-steps
Define Objectives	A. Define broad objectives and constraints B. Estimate quantitative mission needs and requirements
Characterize the Mission	C. Define alternative mission concepts D. Define alternative mission architectures E. Identify system drivers for each F. Characterize mission concepts and architectures
Evaluate the Mission	G. Identify driving requirements H. Evaluate mission utility I. Define mission concept (baseline)
Define Requirements	J. Define system requirements K. Allocate requirements to system elements

The first step in the SMAD process is to define the broad mission objectives and constraints. Additionally, quantified estimates of how well one wants to achieve the broad mission objectives are developed with respect to the needs, constraints, and technology available. These estimates become initial system requirements. A unique feature of the SMAD process is that these quantified estimates are subject to trades as the process continues. Characterizing the mission involves a number of steps. These steps include defining alternative mission concepts and architectures, identifying system drivers for each alternative, and describing in detail what the system is and what it does. Power, weight, and pointing budgets are developed in this step. Evaluating the mission forces the designer to return to the initial system requirements to determine which requirements become driving requirements. Driving requirements are the items principally responsible for determining the cost and level of complexity of the system. Mission utility analysis is also part of this step and this analysis quantifies how well the satellite design meets the system requirements and objectives as a function of design choices. Evaluation of the mission ends by choosing a baseline system design. The SMAD process ends by defining requirements. Broad objectives and constraints are translated into well-defined, specific system requirements. These numerical requirements are allocated to specific components of the overall space mission (Wertz and Larson, 1992:3-90).

The traditional approaches are not suited to designing a satellite bus that will support a variety of missions. This was recognized as the study evolved and initial iterations of the applied processes failed to narrow the scope of the study. Specifically, Hall's approach does not provide an effective, streamlined method for converging on viable satellite design alternatives. Time is wasted performing numerous iterations of the process to achieve the desired focus. Likewise, the SMAD process concentrates too much on using the satellite's payload (mission module) as the key upon which the satellite bus is designed. Consequently, neither of these methods is adequate for designing a generic, standardized satellite bus. A new, customized approach was developed that permitted the team to converge quickly on a satellite bus design without regard to a particular mission module type. The systematic process that was created is a synthesis of the methods described by Hall and SMAD. The process is called the Modsat approach.

Table 2-2: Modsat Systems Approach

Step	Action
Problem Definition	Scope nature of problem
Value System Design	Capture decision maker's needs and goals; create evaluation structure for alternatives
Trade Studies	Link broad design decisions directly to the study's goals and objectives
Modeling	Formulate predictive or descriptive tool(s) to represent activities; analyze various configurations
System Synthesis	Create alternative solution sets
System Analysis	Score each alternative against problem's evaluation structure
Decision Making	Perform sensitivity analysis on solution sets
Implementation	Develop plans for fielding the selected alternative(s)

The iterative approach is comprised of eight steps. The steps, in order, are problem definition, value system design, trade studies, modeling, system synthesis, systems analysis, decision-making, and implementation. The majority of these come directly from Hall's seven-step process. The items that distinguish this approach from Hall's approach are the inclusion of a trade studies step and the reordering of the system synthesis and modeling steps. Additionally, the design team's approach does not include an optimization step. This process distinguishes itself from the SMAD process in two ways. The systematic approach does not commit its focus to the requirements of one mission module as the key factor for satellite bus design. Secondly, the process specifically includes a method for evaluating the merits of each design alternative.

Problem definition is a fundamental first step of any systematic process. The Modsat problem definition step closely follows that of Hall. The purpose of this step is to define and constrain the problem. A result of this step is a succinct statement that identifies the goal and focus of the study. The value of the problem definition step is that it serves as a mechanism to define the system boundaries, identify the system needs, alterables and constraints, and to identify the system actors.

The system boundaries define the environment affecting the system. A distinction can be made between those items contained within an internal environment and those items contained in the external environment. Items within the internal environment are factors that the design team can control. Items that exist in the external environment influence the study but cannot be controlled by the design team. The distinction between the internal and external environment is paramount to understanding the scope and focus of the project. The focus of the study can be narrowed further by performing iterations on the system boundaries. Needs are the fundamental requirements that the decision-maker and users levy on the system and are crucial in determining the broad objective of the study. As is the case in the SMAD process, some needs serve as driving requirements for satellite bus designs. Other needs may be traded off against each other. Alterables are those items that can be influenced or changed by the design team and are contained within the internal environment of the system's boundaries. Constraints are those items that the team cannot control but have a major impact on focusing the study. Problem definition also identifies the actors in the study. Actors are simply the persons/groups who influence the design and evaluation of possible alternatives. Different tools such as concept maps, waterfall diagrams, and interaction matrices can be used to assist in defining the system boundaries, needs, alterables, constraints, and actors.

The value system design step is similar to Hall's respective step. The purpose of this step is to capture the decision-maker's values and goals. Ultimately, these values and goals are used as a means for evaluating the effectiveness of design alternatives. Capturing the decision-maker's values and goals is accomplished by creating an objective hierarchy. Broad values and goals are translated into broad objectives. The broad objectives are decomposed into more specific subobjectives until meaningful measures of effectiveness can be determined. The study's objectives and subobjectives are related to the needs, alterables, and constraints defined in the previous step. As part of defining the study's objectives and measures of effectiveness, major premises and assumptions are explicitly articulated.

Once the objective hierarchy is in place the decision-maker's preferences for each objective have to be incorporated into the structure. It is common to have competing objectives for a problem or study. A score, or weight, is assigned to each objective per level in the hierarchy to capture the importance the decision-maker places on a particular objective. The weights are normalized and the resulting weighted objective hierarchy eventually serves as the evaluation structure for each solution alternative generated in the problem.

The trade studies step is a new and innovative step. This step evolved out of the SMAD process. The purpose of the trade studies step is to make broad design decisions that can be directly linked to the study's goals and objectives. The emphasis is on decisions which can be made without having detailed descriptions of the alternatives. Trade studies serve as an efficient and effective means to narrow the study's scope and provide clearer focus early in the design process. The step is efficient because a manageable study focus can be reached without the need for extra iterations of the process. The step is effective because it reduces the number of possible design alternatives that would have to be evaluated to determine a solution to the study.

The trade studies occur on two levels: the system level and the subsystem level. Trades performed on the system level have broader effects on the design of a satellite bus. These system level trades add definition to the external environment by providing constraints on the system's boundaries. System level trade decisions also impact the trades performed on the subsystem level.

A satellite bus is a system comprised of smaller subsystems. Each subsystem can be designed in a variety of configurations using different qualities and types of components. Some choices can be made independent of choices in other subsystems. A subsystem design decision that is traceable to the study's goals and objectives increases the possibility that system design alternatives will meet the goals of the study. Defining a subsystem configuration or specifying a particular quality or type of component reduces the number of iterations a designer may have to perform to create a viable design alternative. An additional benefit of including a trade studies step early in the process is that it forces team members to focus efforts on gaining insight into subsystem design while simultaneously refining the problem.

Although the trade studies step evolved from SMAD, it differs from the SMAD process in two ways. System level trades in SMAD occur late in the process (the "evaluate the mission" step). This results in the study's focus and system boundaries not being fully defined until late in the process. Subsequently, time is wasted early in the process by identifying the principal cost and performance drivers for each mission concept and mission architectures alternative before the system's boundaries are defined. Secondly, SMAD does not specifically mention that subsystem level trades would occur in the process. It can be inferred that the subsystem level trades would occur after the baseline concept is determined.

The next step in the approach is modeling. Modeling is the development of a descriptive or predictive model representing a set of activities or the entire system in order to allow analysis of alternative configurations of the system (Mosard, 1982:86). The modeling step precedes the system synthesis step, unlike Hall's traditional approach. This reordering of process steps is because the creation and development of satellite bus design alternatives is tedious and complex. Satellite design is an art because many satellite components have to be strategically placed within the confines of a satellite structure to meet stringent heat dissipation, thermal shielding, center of mass, volume, mass, and size constraints. Modeling provides a tool that permits the three-dimensional visualization of the placement and performance of components. Components can be placed, moved, and resized quite easily using a model when compared to physically connecting, disconnecting, or replacing components on an actual satellite. Time and cost savings can be easily realized through the use of a model, especially if design requirements or assumptions change.

In addition, the model builder can take advantage of the decisions that have already been accomplished during the process. Desired subsystem configurations and component selection from the trade studies step can be easily loaded into the model before alternatives are created. If component selection changes, the new information can be easily loaded into the model. Modeling must also be able to quantify the performance characteristics of each design alternative. Different subsystem characteristics can be emulated using mathematical models that can be programmed into the tool. The team's modeling section currently uses the first order estimates and relationships that are found in the SMAD process. Refinements to these relationships can be loaded into the model as the design develops. As a minimum, the quantified performance values must be those values necessary for input into the value system's measures of effectiveness.

The model must allow analysis of alternative configurations of the system. The evaluation structure developed in the value system design is incorporated into this stage of the process. This puts an evaluation structure in place before any design alternatives are generated. With effective use of the modeling tool, it is possible to create new designs and perform evaluation on those designs in a timely fashion.

The system synthesis step is similar to most systematic approach steps for creating alternatives. Accordingly, alternatives can be existing designs, modifications to existing designs, prepackaged designs, or entirely new designs (Pohl, 1995). The difference between the system synthesis step and traditional steps is its placement after the modeling step, for the reasons discussed above.

The systems analysis step follows system synthesis. The purpose of systems analysis is to score each of the design alternatives against the problem's evaluation structure. The problem has been defined and the weighted objective hierarchy is in place.

Each alternative's input to the objective hierarchy's measures of effectiveness is evaluated and each solution alternative receives a score commensurate with its performance to the competing objectives.

Decision-making is a step that permits the team to perform sensitivity analysis on the design alternatives. Sensitivity analysis is performed by varying one variable at a time. This variable is usually a weight associated with an objective in the objective hierarchy. The results of the sensitivity analysis provide insight as to how an alternative will perform given different preferences of the decision-maker. Including the sensitivity analysis results allows the decision-maker to make a subjective decision as to which design alternative will meet the goals of the study.

Implementation is the final step of the systematic approach. The purpose of this step is to develop plans for fielding the selected alternative. The plan is presented to the decision-maker and reflects the team's view on how the alternative can be best put to use in the operational setting. It provides recommendations for improvements to the selected alternative and to the associated elements that affect the alternative. The implementation step also addresses the possible architectures in which the alternative can be deployed and it covers the organizational structure necessary to support that architecture.

The approach described above is an innovative approach to satellite bus design. It provides a logical sequence which deliberately allows the design to evolve from one stage to another while documenting the decisions and assumptions made along the way. The method permits continual improvements to the design as the design matures. This approach provides a method for determining if a design alternative is the best design possible by incorporating design decisions made throughout the process. The systematic approach used in this design study provides a holistic view of the problem and allows the team to capture all important aspects affecting the design. This iterative, systematic approach ensures that these aspects are correctly integrated throughout the design process.

3. Team Organization

This combined approach permitted the team to be easily divided into major areas of responsibility within a matrix organization. The team concentrated on three areas: the major steps of the combined Hall/SMAD systematic approach, particular satellite subsystems, and specific areas of research. Each team member's responsibility included taking the lead in charting the group's direction for the steps of the Hall/SMAD (reference Table 3-1) while maintaining a focus on the team's limited time, resources, and budget. Decisions made in one area of the design or a step in the process had to be properly documented and presented to the group to prevent conflicts between satellite subsystems and maintain the direction of the project. Team members also provided the group with the information necessary to understand each satellite subsystem and to realize the influence and impact each subsystem had on the other. The subsystem assignments are listed in Table 3-2. Each member also made contacts with aerospace companies or organizations that had been involved with the development of satellites within the project's weight class (see Table 3-3). Extremely valuable information was gained by examining the successes and failures of other organizations. The following three tables depict the structure of the team's matrix organization.

Table 3-1: System Steps Responsibility Matrix

Steps Of Hall/SMAD Approach	Member(s) Responsible
Problem Definition	From/Krueger
Value System Design	Cokuysal
Trade Studies	All
Modeling/Analysis	Carneal/Ashby
System Synthesis	Buck
Systems Analysis	Carneal/Ashby
Decision Making	Donmez
Implementation	Donmez

NOTE: Robinson served as a "floater" throughout the Hall/SMAD approach.

Table 3-2: Subsystem Expertise Responsibility Matrix

Subsystem Area	Member(s) Responsible
Structures/Mechanisms and Thermal Control	Ashby
Electrical Power Generation and Distribution	Krueger
Attitude Determination and Control	Robinson
Propulsion	Cokuysal
Telemetry, Tracking, and Commanding/Data Handling	Carneal/From
Mission Modules	Buck
Launch Systems/Command, Control and Communications/Operations Concepts	Donmez

Table 3-3: Similar Projects Research Responsibility

Research Area	Member(s) Responsible
Spectrum Astro/MSTI	Cokuysal
TRW and CTA/STEP	Ashby
Lockheed-Martin/Iridium	Carneal
Orbital Sciences Corporation/Pegasus	Donmez
AeroAstro/HETE	From
Ball Aerospace	Buck
Naval Research Laboratory\Clementine	Robinson
Phillips Laboratory/MightySat	Krueger

PHASE I

4. Introduction

Phase I represents the design team's initial efforts at applying a systematic approach to the preliminary design of a generic satellite bus for small tactical satellite applications. This phase is included to document the design team's progression from the start of the project to the end. An important note is that many changes were made at the end of this phase that resulted in the creation of a new systematic approach that was applied in the subsequent phase of the study.

The team originally tried to apply Hall's seven step process to the study while attempting to incorporate the methodologies described in the Space Mission Analysis and Design text. The flow of this section follows the traditional steps included in Hall's process. This was an important phase for the team because it served as an introduction to the problem and to methods used to solve problems. Each team member gained experience through learning how to apply the theories and methods developed in the classroom. Perhaps the greatest experience, and benefit gained through this process, was the fact that the team was trying to solve an actual United States Air Force problem.

The intent of this iteration was to establish the nature of the problem and its solution space. Upon fleshing out the objectives of the project, broadly defined alternative architectures were generated which could satisfy these objectives. These alternatives were subjected to a qualitative comparison with each other in terms of how well they met each objective. By applying a systematic approach and performing the required steps of the process, each team member gained insight into the complexities and intricacies involved in the design of a satellite bus. The results of this phase served as a starting point for the second iteration of the overall systematic approach.

4.1 Motivation

Imagine the following scenario in the early 21st Century. The United States is a major global power and multi-regional conflicts continue to be the norm. Although massive Department of Defense funding reductions have left the military with minimal resources, the United States still serves as the United Nation's primary police force. The key to maintaining stability in volatile regions, given reduced resources, is to effectively employ Information Warfare in tactical situations. Through the use of rapidly deployable standardized satellite buses, payloads can be launched to provide the situational awareness necessary to effectively employ Information Warfare.

Once a remote sensing package or other type payload is launched, a theater commander will have access to the satellite information through the use of an in-theater satellite command and control/data processing vehicle. The payload data can be downlinked in near real time to support the theater commander's planning efforts or to make an assessment of actions taken. Paramount to maintaining its ability to be a global stabilizing force, the United States needs to design and employ standardized satellite buses to effectively apply Information Warfare. This scenario provides a background for which the satellite bus will have to be designed.

The use of space assets is becoming increasingly essential to ensure victory on the battlefield. United States space forces must be responsive and flexible in order to support the warfighter's short-notice requirements. Typically, each new mission drives the design of a new satellite from scratch. This results in an untimely response and an expensive program. Time delays and increased cost are experienced in the areas of satellite development, assembly, and launch. This process leads to unique integration requirements for launch.

A significant improvement in cost and responsiveness will be achieved if a given class of mission payloads are placed on a generic standardized satellite bus, pre-configured to mate with a launch vehicle. The current launch vehicles most suitable for tactical applications are the Pegasus and the Lockheed Martin Launch Vehicle. This satellite bus is envisioned to be the backbone of a new generation of rapidly deployed, tactically oriented satellites that will support the United States information needs into the next century.

4.2 Background

The project is sponsored by the United States Air Force's Phillips Laboratory. Similar design studies have been completed by various companies and labs, but to date success has been limited. Phillips Laboratory is seeking a "clean-sheet" approach to the design of a cost-effective satellite bus. Several design characteristics have been suggested and will be considered throughout the project. The sponsor is seeking a design that is modular, flexible, robust, and operable; as defined below. These characteristics have been treated as guidance in developing objectives and alternative mission architectures and are not treated as hard requirements.

- Modular - This concept envisions the satellite bus as a large motherboard where components and sensors can be "plugged-in" like personal computer expansion cards to meet specific system needs.
- Flexible - Along with modularity, the bus might be designed such that additional capabilities such as added memory storage can be inserted or removed as mission requirements dictate.
- Robust - Low cost and ease of manufacturing take precedence over technological optimization. Commercial Off the Shelf (COTS) technology and hardware should be used wherever possible.
- Operable - This satellite system is intended for use by military personnel. Ideally, the bus should be designed such that final modular assembly, system test, and component/sensor removal and replacement activities can be accomplished by an Airman with high school education and technical school training.

The first step of the systematic approach is to define the problem. The next section provides information necessary to understand the scope of the team's approach and to solidify what the team hopes to accomplish in this project.

5. Problem Definition

5.1 Introduction

Problem definition was the first step of the systematic approached. The purpose of the problem definition step was to evaluate the proposed problem and establish a succinct problem statement. Defining the problem required careful examination of the sponsor's tasking statement and the factors influencing the proposed problem. Identification of the system's boundary, needs, alterables, constraints, and actors were important to understanding the scope of the problem.

The system's boundary defined those elements of the problem, and its potential solution space, that could or could not be controlled or manipulated by the design team (Athey, 1992:13). Through careful examination and identification of the problem's boundary, the design team determined the factors that influence and affected the problem.

Needs were the driving factors behind the existence of the problem. Needs were referred to as requirements. Without the needs, there would have been no problem. By identifying the needs of the chief decision maker (CDM), the team understood why the problem existed, what the problem was, and what some of the possible solutions to the problem were. Needs also served as a means for measuring the success of potential solutions.

Alterables were those factors the CDM had control over. Identifying those factors provided the team with a method of opening the potential solution space to the problem.

Constraints, on the other hand, were factors that the CDM and design team had no control over. These factors limited the number of potential solutions to the problem.

Actors were the people who had an influence on the problem and the possible alternatives. The most influential actor was the chief decision maker. Capturing and incorporating the decision maker's needs, values, and constraints was paramount to producing the best solution possible. The decision maker provided information necessary to determine the framework by which all possible alternatives were measured.

Problem definition was iterative in nature and the resulting problem statement was subject to change with future iterations. This section of the report provides the design team's "first cut" at the identification of the system's boundaries, needs (requirements), alterables, constraints, and actors.

5.2 Problem Statement

The team's first definition of the sponsor's problem was:

Design a rapidly deployable, tactically oriented, modular satellite bus to enhance theater operations. This satellite bus is to support missions in the Pegasus and Lockheed-Martin Launch Vehicle (LMLV) weight class.

5.3 Concept Map

A concept map was employed to help define the problem. The concept map provided a graphic representation of the design team's interpretation of the problem. Concept mapping is based on the premise that all knowledge can be represented by relationships between more fundamental concepts (Kramer,1990:652-654). The concept map consists of two primitives: concepts and linkages. As an example, refer to Figure 5-1. The satellite bus is the central concept. The motherboard architecture is another concept. The device which connects the two is the linkage. The linkage in this case is "has a". This method ties the two concepts together into a meaningful structure.

The team developed the concept map in Figure 5-1 by carefully evaluating the concepts and linkages suggested through the chief decision maker's tasking statement. This graphic represented the design team's interpretation of the decision maker's problem.

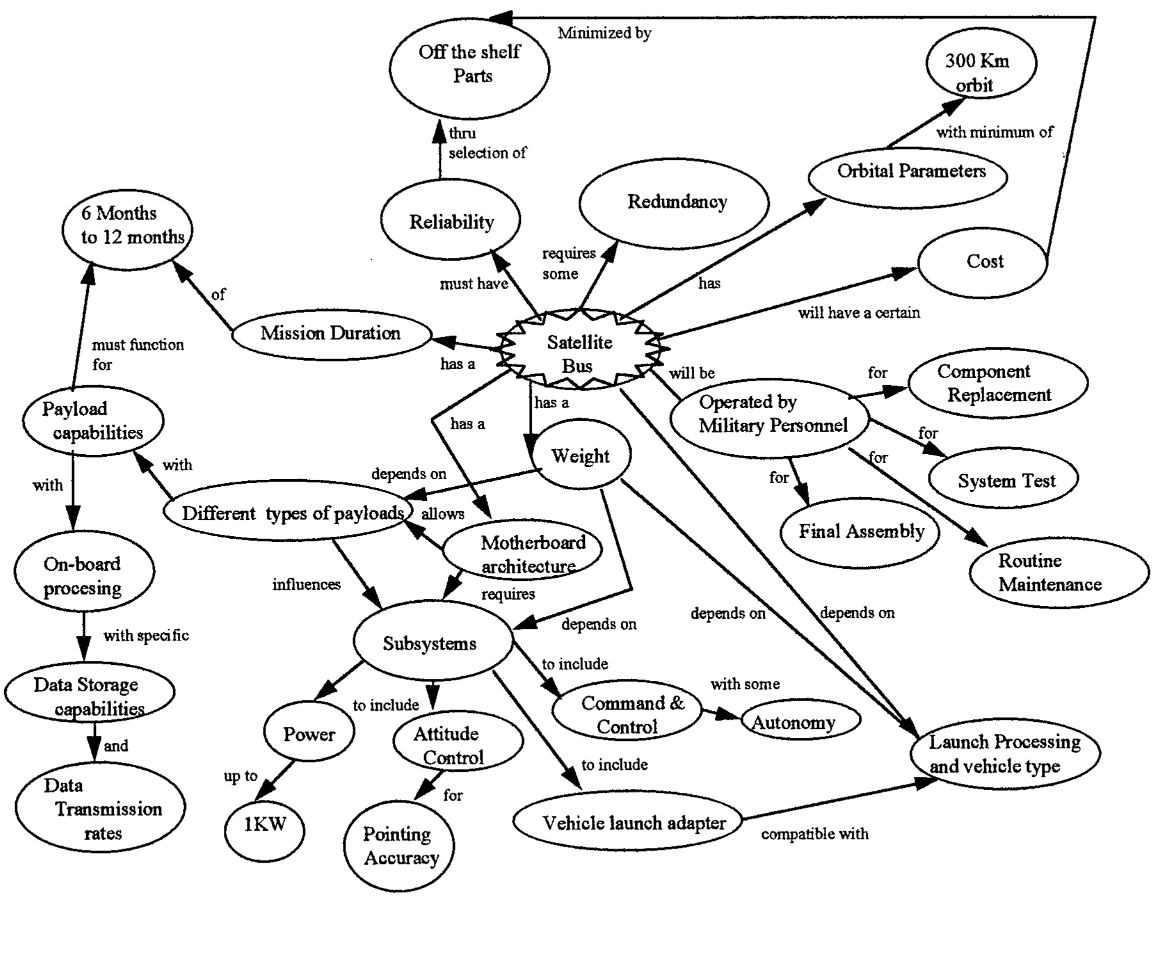


Figure 5-1: Concept Map of Problem

The use of the concept map provided many benefits. It helped the team understand how various factors affected the problem. The team immediately realized that the problem was highly complex. The concept map generated many questions the team needed answered before a concentrated approach to the solution could be given. The team began to question how the operations concept affected a satellite design. How would integration and launch processing occur? Was launch vehicle selection an area to be explored? Another question centered on what type of components were "off-the-shelf" and what reliability did they have. The team also wanted to know what effect orbit selection might have on vehicle life time.

The concept map helped the team identify issues that needed to be considered when examining candidate solutions. The team began to question how much modularity is needed in a satellite bus design and whether modularity is necessarily good. Other questions focused on how much autonomy a satellite bus needs and what reliability is required for a one year life time.

The use of the concept map was only a starting point. The team realized that much research was needed to fully understand the problem. Questions prompted through the use of the concept map were instrumental in identifying areas where research needed to be performed. These areas included researching similar projects, satellite subsystems, launch vehicles, satellite design concepts, command, communication and control architectures, orbital mechanics, and potential mission modules. The concept map offered yet another benefit. This representation of the problem provided a potential mechanism for the team and the decision maker to fully discuss what the problem was and what it was not. The definition of the problem was further enhanced by establishing system boundaries.

5.4 System Boundary

The system boundary defined those elements of the problem and its potential solution space that could be manipulated by the design team. Anything outside of the boundary was considered outside the team's scope. By determining the items that resided inside the system boundaries, the team was able to focus the problem into a more manageable size.

As a first attempt at scoping the problem, it was decided that the satellite bus subsystems and interfaces, as well as mission module interfaces, were within the system's boundary. Operational concepts directly related to the use of the bus were also within the boundary. These concepts included on-orbit command and control, storage and maintenance, mission module and launch integration, and sensor data processing. Items that were not considered inside the system's boundary included the development of specific mission modules, launch vehicles, and command and control facilities. It was also determined that the team would not concentrate on innovative or new state-of-the-art technologies.

5.5 Needs

Understanding the chief decision maker's needs were an integral part of defining the problem. Using the sponsor's tasking statement as a guide, the team tried to capture the CDM's needs and categorized these needs into four functional areas; performance, responsiveness, operational, and development. The following provides a list of the sponsor's needs and shows how the needs were categorized.

- Performance

  - Pegasus/LMLV weight class
  - Modular, interchangeable components
  - Insertion or removal of capability as needed
  - Support various sensors (i.e., electro-optical, IR, laser, SAR)
  - 1 meter Electro-Optical (EO) resolution for a 5-10 minute imagery pass
  - 1 meter Synthetic Aperture Radar (SAR) resolution, 10 kilometer by 10 kilometer imaging
  - Support on-board storage of up to 100 SAR images
  - Support minimal on-board processing and data compression algorithms
  - Near real-time transmission of 1 m resolution SAR data
  - Encryption of data

- Responsiveness

  - Tactically oriented (had to support tactical mission modules)
  - Rapidly deployable

- Operational

  - orbit equal to or greater than 300 kilometers
  - up to 12 month mean mission duration
  - Command uplink/telemetry downlink compatible with the Air Force Satellite Control Network

- Development

  - Ease of manufacturing
  - Off-the-shelf technology
  - Avoid use of Class-S parts where possible
  - Avoid extensive redundancy

These lists helped identify what the sponsor was trying to accomplish. The team examined these needs and tried to determine if any of the needs conflicted with each other or could not be met. The major concern was whether a satellite bus could be designed to support the specific number and types of mission modules the sponsor required. The team concluded that more information was needed before a decision could be made to eliminate any of the needs from the scope of the problem. For this iteration, the team would attempt to meet all the needs of the sponsor.

5.6 Alterables and Constraints

Alterables were those elements of the system and its environment that could be controlled by the chief decision maker. The constraints were those items which could not be changed by the decision maker. The team had to manipulate the alterables to achieve a solution, provided the constraints were satisfied. For this iteration of the project, it was determined that every aspect of the satellite bus was alterable. This offered a great deal of flexibility in the design of bus subsystems and payload interfaces. For the first iteration, the satellite orbit would also be left as a variable. The satellite could be launched into a range of altitudes within Low Earth Orbit (LEO) by an appropriate small launch vehicle. Additionally, a variety of operational concepts were considered. These concepts included areas such as bus storage and processing, payload integration, launch integration, command and control, and data processing and distribution.

Table 5-4 shows the type of questions that were considered when the alterables were examined (Wertz and Larson, 1992:21).

Table 5-4: Elements of the Mission Concept of Operations

Data Delivery	How is the mission data processed and delivered to the end-user? How much processing is done on-board?
Tasking, Schedule, and Control	How autonomous is the system? What automatic housekeeping functions will be provided? Fixed or mobile (or both) command and control?
Communications Architecture	How is sensor data returned to Earth?
Mission Timelines	When will first bus be integrated and launched? Time from placement of order to launch? (integration, assembly, test, launch processing)

Answers to these questions were left to the team members who were performing research in the appropriate areas.

5.7 Actors

The actors were all the people and agencies who were involved with some aspect of the system or project. It was important to understand and consider the impact the system has on all actors. The principle actor for any project is the chief decision maker (CDM). The CDM generates the requirements and objectives for the system, and is the approving and implementing authority for the solution. The CDM for this project was LtCol James Rooney of Phillips Laboratory, Kirtland Air Force Base. The design team was comprised of the engineers and analysts who will work together to develop the system. This project's design team consisted of space operations and systems engineering masters candidates at the Air Force Institute of Technology (AFIT). The team was advised by members of AFIT's Department of Aeronautics/Astronautics.

The eventual user of bus design was also an actor in this design study. This was the warfighter who depended on tactical space assets to wage effective information warfare. In order for the warfighter to receive his information, the project's satellites would be commanded and controlled by Air Force space operators. Air Force launch personnel would integrate the satellites to launch vehicles and launch them into low earth orbits. Prior to launch, mission modules would be integrated with their satellite busses by qualified personnel. Prior to being required for missions, ready-to-integrate busses would be stored and maintained. All personnel required to complete these activities were important actors in the development of this system, and their needs were considered.

6. Value System Design

6.1 Overview

A systems engineering approach to the development of a system considers the values and objectives of the chief decision maker (CDM). The requirements and values expressed by the CDM must be expressed as an organized set of system objectives. This set of objectives should drive all design efforts, and it must serve as the standard by which alternative solutions are evaluated. Often, the established objectives are in conflict; that is, positive performance for one objective may imply negative performance for another. An example of this is the use of cutting-edge technology, which may deliver high performance while admitting higher cost and technological risk. The objectives must be organized in such a way that the engineer may judge alternative solutions against all the objectives, and perform trade-off analyses where necessary.

Value system design translates CDM values into a hierarchy of objectives, where objectives flow down from the top level in a well-structured manner. Each bottom-level objective has a corresponding measure of effectiveness (MOE), by which the performance of that objective is measured. In addition, each objective is weighted, in terms of importance, relative to the other objectives at the same level. Based on the performance of each objective, and the priorities of those objectives, alternative solutions receive a utility score, which can be compared to the scores of other alternatives. In this way, the value system allows the engineer to analyze tradeoffs between competing objectives.

When discussing the objectives for this study, it is very important to remember the system boundary, that is, what is within the scope of the study. Many of the objectives touch on areas of space and launch operations. While it was not the intent of this study to analyze these areas for design and improvement, the design of Modsat will definitely impact the areas of space and launch operations. It is this impact that must be considered.

It is also important to consider the timeframe within which the elements of this study are relevant. For instance, some of the objectives apply to the development of the bus, others apply to the assembly and integration of the satellite, and still others apply to the operation of the satellite. The timeframe for each objective should be clear from the context of that objective.

6.2 Objectives

The team determined that the overall objective of this study is to:

"DEVELOP THE BEST STANDARDIZED BUS FOR A SMALL TACTICAL SATELLITE."

It is important to understand what the words in this objective mean in order to prevent any misinterpretations between team members, advisors and the chief decision maker. The key words are defined below:

1. STANDARDIZED: The most important idea of the study. All other objectives must support this. The idea is that the bus will support many different mission types.
2. THE BEST: Implies an open-minded approach to developing the best system, free from the bias of favored technologies and approaches.
3. SMALL: The satellite must be compatible with a light weight launch vehicle, such as the Pegasus or the Lockheed Martin Launch Vehicle.

4. TACTICAL: The project statement emphasizes tactical missions and a short life.

All objectives for this study were derived to fit all top level mission requirements, as defined in the problem definition step. As with any project, there is a trade-off between cost and effectiveness. These are important aspects of this design study, and warrant much consideration. Therefore, these two conflicting objectives became the main top-level objectives. The top-level objectives are shown in Figure 6-1.

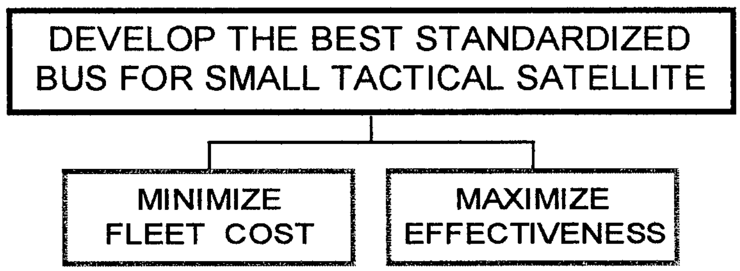


Figure 6-1: Top-level Objectives

6.2.1 Minimize Fleet Cost

Cost is the common value of worth for all parts, processes, actions, and decisions. Without careful consideration in separating the elements of cost, areas of overlap can lead to double counting, which inflates predicted system cost. It should be noted that launch costs are beyond the scope of this study. As this standard satellite bus will be designed for multiple applications, the cost for fleet production has been considered. Cost objectives are shown in Figure 6-2.

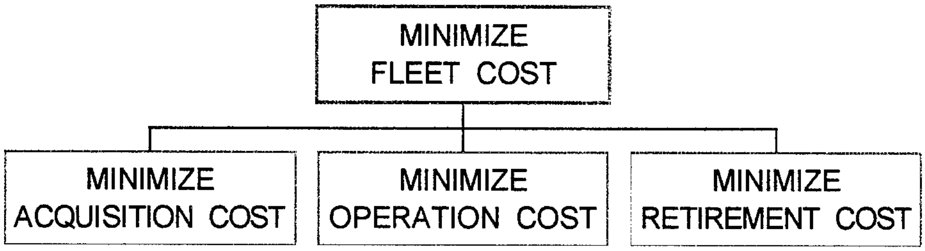


Figure 6-2: Cost Objectives

6.2.1.1 Minimize Acquisition Cost

Acquisition costs cover all recurring and non-recurring costs as the system progresses from an idea to a finished end item.

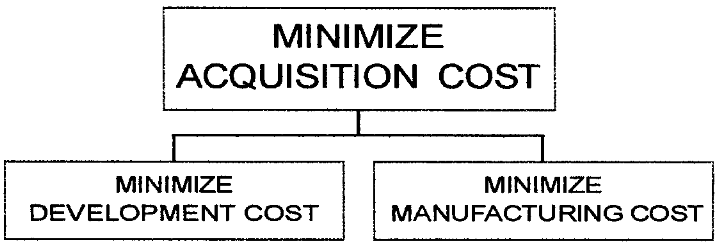


Figure 6-3: Acquisition Cost Objectives

O-1. <u>Minimize Development Cost:</u> Includes all expenditures from problem definition to production.

O-2. <u>Minimize Manufacturing Cost:</u> Cost of producing the system; includes qualification testing.

6.2.1.2 Minimize Operations Cost

The operations phase begins with the satellite as a finished product, and includes integration with the payload and launch system. It also includes on-orbit operational costs such as command and control, data processing and other ground segment expenditures. Operations cost objectives are shown in Figure 6-4.

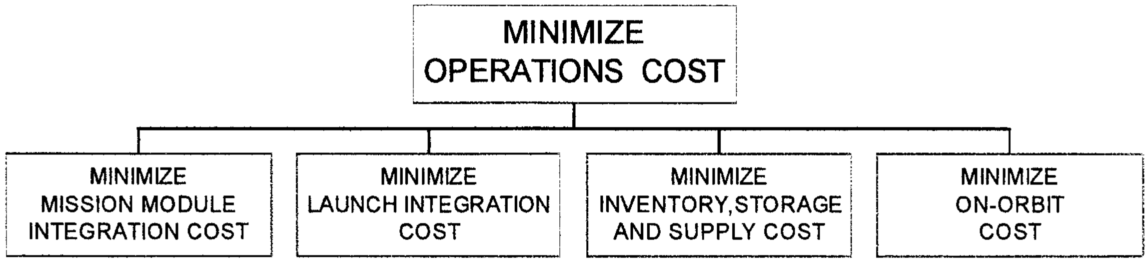


Figure 6-4: Operations Cost Objectives

O-3. Minimize Mission Module Integration Cost: The cost of mating a mission module to the bus; covers all hardware and software interface and testing expenditures.

O-4. Minimize Launch Integration Cost: Covers all efforts to fit the mission-ready satellite to a launch vehicle.

O-5. Minimize Inventory, Storage and Supply Cost: The cost of having a fleet of satellite busses in a mission-capable configuration . Between manufacturing and usage, the fleet must be stored, cared for, and kept mission-ready.

O-6. Minimize On-Orbit Cost: All expenditures for orbital/system maintenance such as command and control, data processing, and ground segment activities.

6.2.1.3 Minimize Retirement Cost

O-7. Minimize Retirement Cost: Upon conclusion of a satellite's operational life, environmental and orbital space considerations deem that it be removed from operational orbit. This cost must be considered up front.

6.2.2 Maximize System Effectiveness

The elements which define system effectiveness have been chosen as shown in Figure 6-5.

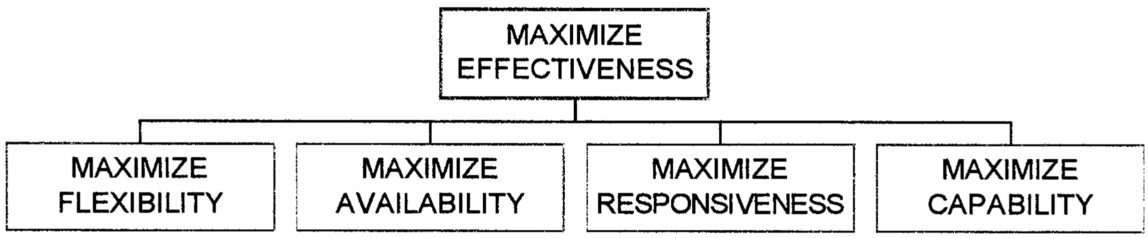


Figure 6-5: Effectiveness Objective

6.2.2.1 Maximize Flexibility

The standard bus must be flexible for use with different missions, payloads, launch vehicles, orbits and other specifications. Flexibility has been defined as shown in Figure 6-6.

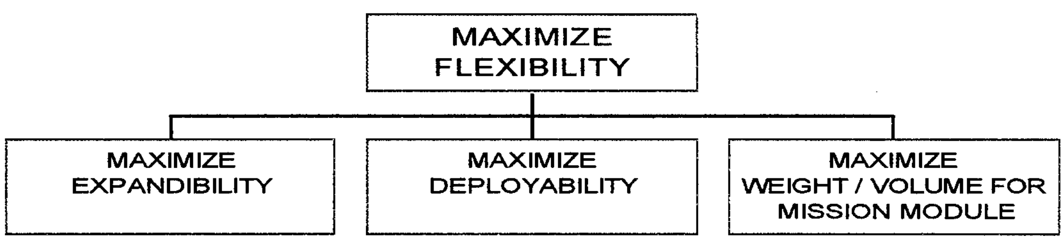


Figure 6-6: Flexibility Objectives

O-8. Maximize Expandability: The system must respond to different mission requirements while satisfying the constraints. Elements such as power, memory, and earth coverage must be adaptable to different demands.

O-9. Maximize Deployability: Considers the flexibility of selection of orbital parameters for low earth orbits.

O-10. Maximize Weight and Volume For Mission Module: Optimum mission module capacity occurs with minimum bus weight and volume, since total satellite weight and volume are constrained by the launch vehicle.

6.2.2.2 Maximize Availability

The on-orbit availability of the satellite must be maximized, considering the elements shown in Figure 6-7.

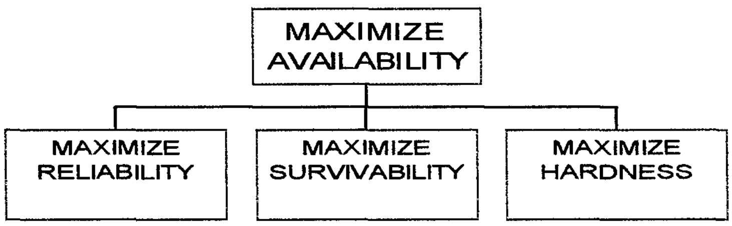


Figure 6-7: Availability Objectives

O-11. Maximize Reliability: Minimize the probability of failure for the overall system over the mission life, under nominal operating conditions (no environmental or man-made threats).

O-12. Maximize Survivability: The ability of the system to perform its intended function after being exposed to stressing environments created by an enemy or hostile agent.

O-13. Maximize Hardness: An attribute defining the spacecraft's ability to withstand natural environmental threats and stresses.

6.2.2.3 Maximize Responsiveness

Since this spacecraft bus is to be used for tactical missions, the time required to meet the warfighter's needs must be minimized. Responsiveness implies a light weight, easily deployable, agile bus that can meet short-term military mission requirements.

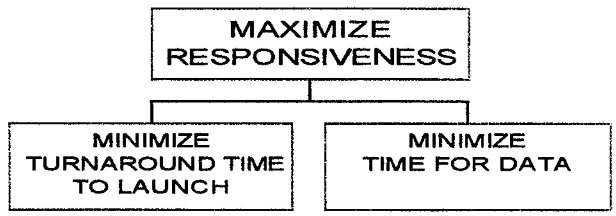


Figure 6-8: Responsiveness Objectives

O-14. Minimize Turn Around Time To Launch: The time period from the initial mission demand until placement in orbit. This time covers hardware and software preparation, mission module integration, and launch integration.

O-15. Minimize Time For Useful Data: The time period from satellite sensor collection of raw data until the warfighter receives his information in the field. Considers queuing, processing, and distribution.

6.2.2.4 Maximize Capability

O-16. Maximize Capability: The system should perform so as to enhance information flow into combat theaters. This topic covers the performance of system elements. It is not restricted to satellite specific elements, but also includes mission module, user, and operator interfaces. The bus should be designed to optimize satellite/user interfaces such as data rates, accessibility, compatibility, and encryption, while satisfying all other objectives as well. On-orbit performance factors include pointing accuracy, thermal control, satellite autonomy, attitude control, and self-diagnostics (Reeves, 1992:285-337). Simplicity of operations should be considered, including mission planning, compatibility of data links, and telemetry analysis.

Note: This objective was created as a the head of several potential performance subobjectives, to be identified in a later iteration of the study. However, in the first phase of the study, it was decided that the first order solutions would not have enough detail to warrant further definition of this objective.

6.3 Measures Of Effectiveness

Each of the 16 bottom-level sub-objectives requires a measure of effectiveness. Objective MOEs based on natural performance scales were not feasible for this first, quick iteration through the design process. Since the purpose of the first iteration is to get a rough idea of what possible alternatives are feasible for this design, the team decided to use subjective judgments to determine the performance of each alternative with regard to the objectives.

Moreover, rather than attempt to find fixed reference points for the performance of each objective, the team decided to judge the relative performance of each alternative with respect to the other alternatives. For each bottom-level objective, team members performed pairwise comparisons of the alternatives, in much the same manner by which the objective weights were determined (see section 6.4). Thus, performance scores were determined for each alternative, objective by objective.

6.4 Priority Weighting of The Objectives

As mentioned in section 6.1, the objectives must be given relative importance in the form of weights. Only the objectives on the same level of the hierarchy should be compared. These objective weights must reflect the priorities and values of the decision maker. Thus, participation from the CDM is critical in completing the objective structure. This was accomplished through the use of a preference chart survey, such as that shown in Appendix A for phase two of this study. A preference chart survey requires the respondent to evaluate the relative importance of items in a series of pairwise comparisons (Athey, 1982). Surveys were distributed to the CDM, faculty advisors, team members, and other subject matter experts.

In the survey, a participant has five possible comparison ratings. When objective A is compared to objective B, objective A could be 'much more important than', 'more important than', 'the same as', 'less important than' or 'much less important' than objective B. Each rating has an assigned value: 4.0, 2.0, 1.0, 0.5, and 0.25, respectively. An objective is compared to all the other objectives on the same level of the hierarchy, resulting in a series of values. A relative score for a given objective is obtained by taking the geometric mean of all its comparison values (Pohl, 1995). Finally, the relative scores for each objective are normalized so that they add up to one (see section 13.6 for a description of the additive utility function used for this study). The resulting value for each objective is the weight of that objective, according to the person who completed the survey.

The results of all the surveys were averaged. Feedback from the CDM and from the faculty advisors was given double emphasis. The resulting priority weights are shown in Figure 6-9.

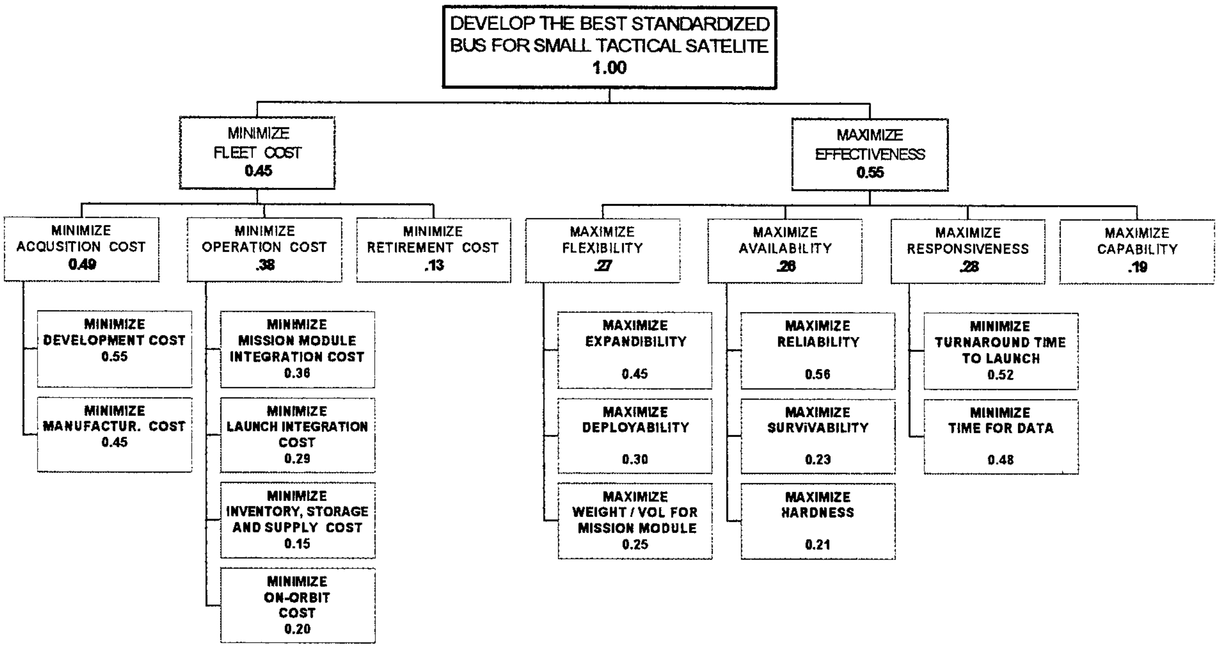


Figure 6-9: Phase One Objective Hierarchy

7. System Synthesis

7.1 Introduction

The goal of system synthesis is to generate alternative solutions which sufficiently span the needs (i.e., meet the minimum requirements) addressed by the problem statement. Given the very general nature of the initial statement of needs, the iterative design approach must start with high-level designs. Through further iterations, these high-level designs should evolve as necessary into more detailed designs, slowly but surely incorporating more elements at lower (component) levels of the system.

7.2 Architectural Themes

The first step of the System Synthesis is to provide top-level alternative architectures which span the myriad of lower-level, detailed choices available for single components or elements of the design. This approach provides an introduction and overview not only to the general (high-level) tradeoffs involved with satellite design, but also to the whole process of generating alternatives for evaluation. The early brainstorming sessions of the system design study provided widely varying ideas spanning all of the major elements of space system architecture. Elements under consideration included not only the spacecraft bus and the payload, but also the command and control infrastructure, the launch and ground segments, etc. (i.e., brainstorming sessions were exhaustive). The brainstorming sessions concluded with the following concepts: [1] care must be exercised in creating initial candidate architectures that are not overly detailed but are differentiable; [2] brainstorming sessions spanned not only spacecraft options but operational and employment options, thus possibly creating multiple combinations too numerous to effectively evaluate as detailed designs in a timely manner; [3] candidate designs must concentrate on the spacecraft element, with options for other space system elements held as secondary; [4] candidate designs must be fully characterized (at the system level -- not the component level) to provide an introduction to and deeper understanding of spacecraft design for those members of the study who are less familiar with satellite technology.

The brainstorming data provided ample ideas for creativity. From the outset of the system synthesis process, it was apparent that a classical "morphological" approach to candidate design generation was inappropriate to the general scope of the team's first (high level) efforts. Therefore, candidate designs known as "architectural themes" developed. This "thematic" approach seeks to span the range of different design philosophies available within the scope of the problem. These themes do not necessarily incorporate different components within the satellite designs they represent, but are intended as differentiable methods for building and employing satellites out of (possibly) the same (i.e., standardized) components. Through this approach to system synthesis, the team can evaluate the different spacecraft architectural themes for their general, relative merits and reserve not only component-level considerations for future system synthesis and analysis, but also other elements, such as employment or command and control options, for future implementation recommendations.

7.2.1 Initial Spacecraft Architectures

7.2.1.1 "Baseline" or "Point" Design

The "Baseline" ("One Size Fits All" design) provides a single bus design capable of supporting any of the required payload types. The "Baseline" supports one payload per flight, and does not incorporate any onboard processing of mission data or spacecraft autonomy. The "Baseline" bus communicates only through the AFSCN. The key philosophy is to design a satellite bus to the most demanding requirements in each subsystem (e.g., power, attitude control, data handling, data storage, etc.) necessary to support any of the required mission options. This design is not optimized for any one particular mission.

7.2.1.2 "Amoebae" Design

The second spacecraft architecture is aptly named the "Amoebae" design. This design, also known as the "Ultimate Chinese Menu," incorporates a fully modular, possibly expandable bus construction, complete with "plug and play" subsystem components such as battery modules, solar arrays, memory modules, reaction wheels, etc. This bus, like the "Baseline", incorporates no special processing, autonomy, or command and control options, but it may support one or more payloads if desired. This design's philosophy concentrates on the ability to tailor and optimize the vehicle to a particular mission, including the possibility of dual or multiple payloads, providing maximum design flexibility.

7.2.1.3 "Family" Design

A possible alternative to the "Baseline" design is one which spans multiple mission or payload needs by providing two or more similar but distinct buses, each designed and built to support a smaller subset of the overall payload types. These buses incorporate no modularity and no onboard processing or autonomy. They support one payload or mission per flight. Although this design philosophy is similar to the "Baseline", the "Family" of satellites will be different in capability, size, mass, and mission applicability. The family of architectures essentially forms a set of "Point" designs.

7.2.1.4 "Microsat" or "Pocketsat" Design

The "Microsat" design architecture provides satellites individually optimized for particular missions (i.e., every mission or payload type has a specific satellite -- no real "standard bus"). This design is not a standard vehicle, although several may coincidentally look and operate similarly. Instead, a "standard" philosophy will be applied across all subsystems incorporating a "seamless" architecture to minimize weight and volume, yet maximize effective performance. Microsats emphasize very small (less than 50 kilograms, possibly man-portable) mass, optimization of payload support requirements, and miniaturization/multifunctionality of components. There will be no discernible "border" between where the bus ends and where the payload begins because all the components on the satellite will have been fully and optimally integrated (for that particular type). Development efforts for minimizing payload mass will mirror those for the bus subsystems. Again, there will be no provisions for autonomy, data processing, or communication other than through the AFSCN.

7.2.1.5 The "Works" Design

The final spacecraft architecture under consideration is the "Works" design. As the name implies, the works spacecraft will include as much capability as possible on one spacecraft bus. This design is intentionally as large as mass constraints allow and is also capable of supporting several payload packages on the same bus. Additionally, the bus incorporates full redundancy, radiation hardening, and satellite autonomy features. Extensive onboard mission data processing is available before downlinking. The bus will have a useful service life of greater than one year. Since many mission features are supported by the software subsystem, the spacecraft is reconfigurable after launch via database uploads. Also, this design is basically a larger, more complex version of the "Baseline" design, enhancing many mission capabilities through sophistication and achieving multiple payload objectives through sheer horsepower.

7.3 Evaluation Considerations

Though very general in nature and deliberately lacking specific components, the design philosophies provide significant differentiation through their approaches to meeting mission objectives. These approaches will drive different factors inherent in each design's technology, cost, operability, reliability, etc. The primary emphasis in each case can be summarized as follows:

1. Baseline Design: support any mission with a single bus
2. Amoebae: support any or more than one mission with a tailorable bus
3. Family: several buses span the desired needs
4. Microsat: each payload is an optimized satellite in itself, minimizing mass
5. Works: large satellite supporting several missions at once

7.3.1 Employment Options

In addition to the core of five spacecraft architectures, some initial operational concepts came to light. Because these employment/deployment schemes did not directly relate to specific satellite architectures, they were considered as implementation recommendations and/or operational alternatives suitable for further investigation.

7.3.1.1 "Fire and Forget"

As implied by the title for this employment/deployment option, operations personnel prepare and launch the spacecraft, which then operates with complete autonomy. The only contact made with the vehicle by operations personnel after launch will be for the collection of fully processed mission data downlinked from the vehicle. The concept of the "probe" vehicle used in certain science fiction stories (probes in the "Star Trek" series are prominent) is a good analogy for this option.

7.3.1.2 "Starnet"

The "Starnet" is a network of crosslinked, autonomous vehicles, creating not only a more robust command and control/data transmission network but also a synergy of the variety of mission data

7.3.1.3 "Reusable" Satellite

This concept utilizes either an in place or deployable heat shield for the de-orbit and recovery of the spacecraft. This employment scheme allows recovery, refurbishment, reloading, and relaunch of a satellite after completion of a particular mission.

8. System Modeling

8.1 Introduction

Discussions up to this point have focused on defining the problem, establishing criteria and MOEs. In the last section the System Synthesis process produced various alternative solutions, which will be investigated further. Typically, in System Synthesis the feasibility of the alternatives are evaluated by checking them against the constraints and verifying that they meet minimum standards. However, because little is known of what the standard should be, all alternatives were passed and evaluated against one another for the 16 objectives discussed in Value System Design. The scores for the five alternatives for each objective were then normalized and these were then evaluated against the remaining upper level objective criteria to obtain an overall rating of that alternative against the others.

The objective was to model the alternatives in such a way as to get results on how well they stack up in the objective hierarchy as established in Value System Design. This process causes the alternatives to have relative worth based upon other ideas in the solution space. Microsoft Excel was chosen as a tool to help automate the process of modeling alternatives in Value System Design.

8.2 Purpose

A tool was needed to model the objective hierarchy created within Value System Design. The intent was to automate evaluation, and to ensure such evaluation was performed the same way each time. The benefits were consistent performance of steps and time saving once the model was created and checked for valid output on calculations.

8.3 Confidence Level in Data

Confidence of the data is limited primarily due to four main factors. First, the data collected from the surveys is only as good as the expertise of those surveyed. Although the data obtained is sufficient for a first look, individuals or groups with more field experience will need to be contacted in future surveys. Second, the data collected for this section was based on only eight inputs, which is clearly insufficient for establishing high confidence tables. Third, although the alternatives were compared equally they will likely vary in many aspects. Manufacturing may be more complex for one design than another, or logistics for storage may be more difficult or costly for one design over another. In essence, it is really difficult to compare a "Microsat" against "Amoebae" without knowing more about the current technology. Lastly, in defining the alternatives assumptions were made which may not be true representations.

In any modeling scenario key assumptions must be made to avoid building too much complexity. Otherwise, the model becomes unmanageable and outside the scope of its true purpose. Because these assumptions have tremendous impact in the outcome of the model, it is important to discuss them here. The following items describe the assumptions for each of the objectives:

Minimize Development Cost:

- "The Works" is fully redundant and autonomous
- Amoebae will be an integration "nightmare"
- Point Design and Family are about equal in development cost
- Microsat technology is revolutionary and will require more money to develop than the other systems.

Minimize Manufacturing Cost:

- "The Works" is an intricate and sophisticated satellite
- Amoebae and "The Works" will require advances in assembly line technology
- The "Family" versus "The Works" is similar to a Yugo versus a Rolls Royce
- Microsat will be easier to build than the other satellites since it will be manufactured much like a circuit board on an assembly line

Minimize Mission Module Integration Cost:

- "The Works" will contain multiple payloads
- Amoebae will be the most complex satellite to try to integrate payloads
- Microsat will be built with the assumption payloads will also be miniaturized

Minimize Launch Integration Cost:

- Amoebae is extremely complex, requiring testing and integration in phases
- Microsat is one payload versus many for "The Works"

Minimize Inventory, Storage, and Supply Cost:

- Amoebae will be more costly because of the increased inventory due to all parts being modular, and parts may change as technology changes
- Family has a small set of fixed designs, and thus has known items to maintain on inventory
- Microsat will require more configuration management
- "The Works" complexity requires more follow-up and hardware/software checkout

Minimize On-Orbit Cost:

- Amoebae is more diverse than the other satellites, thus requiring more resources to maintain it.
- Microsat will be the easiest to maintain since it will have built in self checks
- "The Works" - will work as it was designed to

Minimize Retirement Cost:

- Microsat will be the cheapest
- Because of how much we put into "The Works", it becomes an investment. Its loss will have an impact on development and user community

Maximize Expandability:

- Point Design is fixed and not as flexible as the Amoebae design
- "Family" is a little better than the Point Design
- Amoebae is the best because it is designed with maximum modularity

- "The Works" will be the most difficult to reconfigure once built.

Maximize Deployability:

- Microsat has no maneuvering capability
- "Family" and the Amoebae satellites can by their design carry additional fuel if necessary
- "The Works" is heavy and will require more fuel to maintain orbit parameters than the other satellites

Minimize Weight & Volume For Mission Module:

- Point Design is a larger satellite to accommodate all the various payloads
- "Family" and Amoebae are more tailored to specific payload, thus minimizing weight and volume

Maximize Reliability:

- Amoebae is more likely to break with the many and complex interfaces
- Microsat is newer technology and may not be that reliable in earlier satellites
- "The Works" is built on the premise it will be highly reliable and redundant

Maximize Survivability:

- "The Works" is built for survival against offensive measures
- Microsat is smaller and harder to attack with lasers or ASAT weapons
- Amoebae is not equipped with shielding to maximize modularity

Maximize Hardness:

- "The Works" is built to withstand the harsh space environment
- Amoebae is less likely to survive than the other satellites

Minimize Turn Around Time To Launch:

- "The Works" will require more software/hardware testing than the typical satellite
- Amoebae is the worst, requiring extensive phase testing during integration
- Microsat is the best with a "launch-and-go" ability

Minimize Time For Useful Data:

- "The Works" is slightly better than the other satellites
- Remaining satellites are essentially equal in on-board processing

Maximize Capability:

- "The Works" is slightly better
- Microsat contains payloads that can meet performance requirements

8.4 Evaluation Tool

To evaluate the alternatives an Excel workbook was developed and used. It utilizes weights from the objective hierarchy and accepts inputs from one or more surveys on pairwise comparison of architectural themes. From these it calculates geometric means of each alternative under each sub-objective before normalizing the weights. Next, the model calculates weights of alternatives for each objective, then graphically displays the results.

8.4.1 Survey Model

The eight members of the Systems Engineering Group performed a pairwise comparison of all five alternative architectures for each and every bottom level objective in the objective hierarchy. By using pairwise comparison any given alternative is judged against any other given alternative for a given objective through the following ratings: much better than, better than, equal to, worse than, or much worse than. Criteria used were as follows:

- Alternative j much better than k receives rating "+ +" (numerical equivalent = 4)
- Alternative j better than k receives rating "+" (numerical equivalent = 2)
- Alternative j equal to k receives rating "=" (numerical equivalent = 1)
- Alternative j worse than k receives rating "-" (numerical equivalent = 0.5)
- Alternative j much worse than k receives rating "- -" (numerical equivalent = 0.25)

9. System Evaluation

9.1 Introduction

In the previous section the Team determined a ranking of alternatives one against another, given estimated conditions. The CDM wants to know what happens if conditions of the modeling are different than were expected. The CDM is also interested in knowing what latitude exists if less than the optimal solution is selected. To meet this requirement, conditions of the model need to be modified to provide some flexibility to the original solution set. In the business world this process is called "Decision Analysis," the varying of original conditions to see trends. "Decision analysis is typically an iterative process. Once a model has been built, *sensitivity analysis* is performed. Such analysis answers, 'what if' questions: 'If we make a slight change in one or more aspects of the model, does the optimal solution change?'" (Clemen, 1996:7). System Evaluation uses a similar process, where conditions of a model or criteria are changed to see how alternatives fair under different conditions. One approach is to vary the weights assigned in the overall objective hierarchy. The Excel workbook model would then feed through the new weights and re-calculate the results.

9.2 Pairwise Comparison

Based on results of pairwise comparison, the Microsat option scored with the highest weighting. However, the Microsat solution did not lead the others by a very wide margin, so changes in objective hierarchy weighting may give a different leading option. This result can be seen in Figure 9-1:

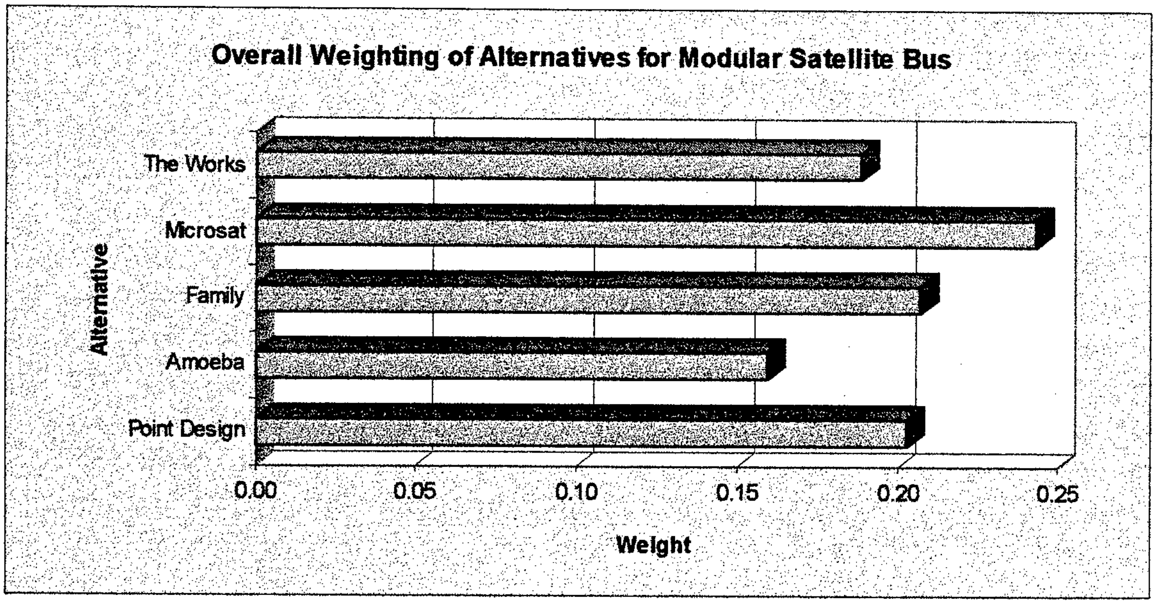


Figure 9-1: Overall Comparison

9.3 Scenarios

To determine if Microsat was the overall best solution, conditions of the evaluation were modified. To do this analysis an individual part of the objective hierarchy was altered significantly by varying the weight to seventy-five percent of its original value in the hierarchy. The other weights at that level were proportionately decreased to maintain the same relative weights, while ensuring the total weights still added to one. An example is shown in

Table 9-1. In this scenario acquisition cost weighting factor was made the most important consideration for fleet cost.

Table 9-1: Acquisition Cost Example for Generating Scenario

Objective	Original Weight in Objective Hierarchy	New Relative Weights in Scenario
Minimize Acquisition Cost	0.49	0.75
Minimize Operation Cost	0.38	0.19
Minimize Retirement Cost	0.13	0.06
TOTAL	1.00	1.00

The two primary scenarios were generated and compared:

1. Fleet cost is the driving factor

2. Maximizing effectiveness is the driving factor

To determine the impacts of subobjectives on the solution set, seven sub-scenarios were generated and compared:

1. Acquisition cost is the most important consideration for fleet cost

2. Operation cost is the most important consideration for fleet cost

3. Retirement cost is the most important consideration for fleet cost

4. Flexibility is the most important consideration for effectiveness

5. Availability is the most important consideration for effectiveness

6. Responsiveness is the most important consideration for effectiveness

7. Capability is the most important consideration for effectiveness

The Excel model was used as the tool to perform the above scenarios and generated results shown in the following figures:

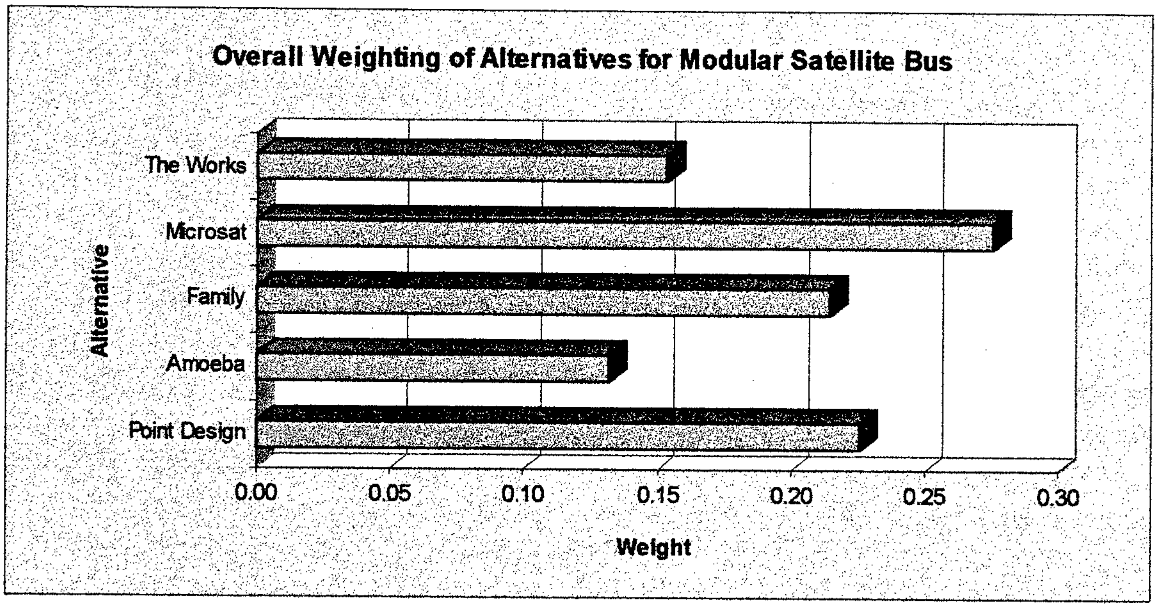


Figure 9-2: Scenario Results for *Cost* as the Driving Factor

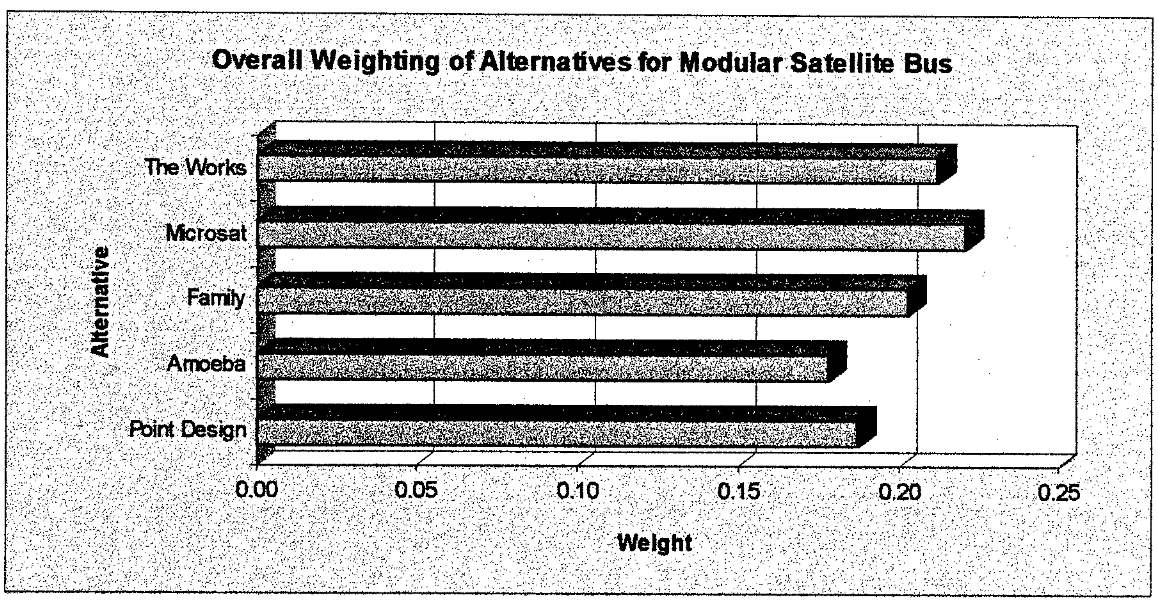


Figure 9-3: Scenario Results for *Effectiveness* as the Driving Factor

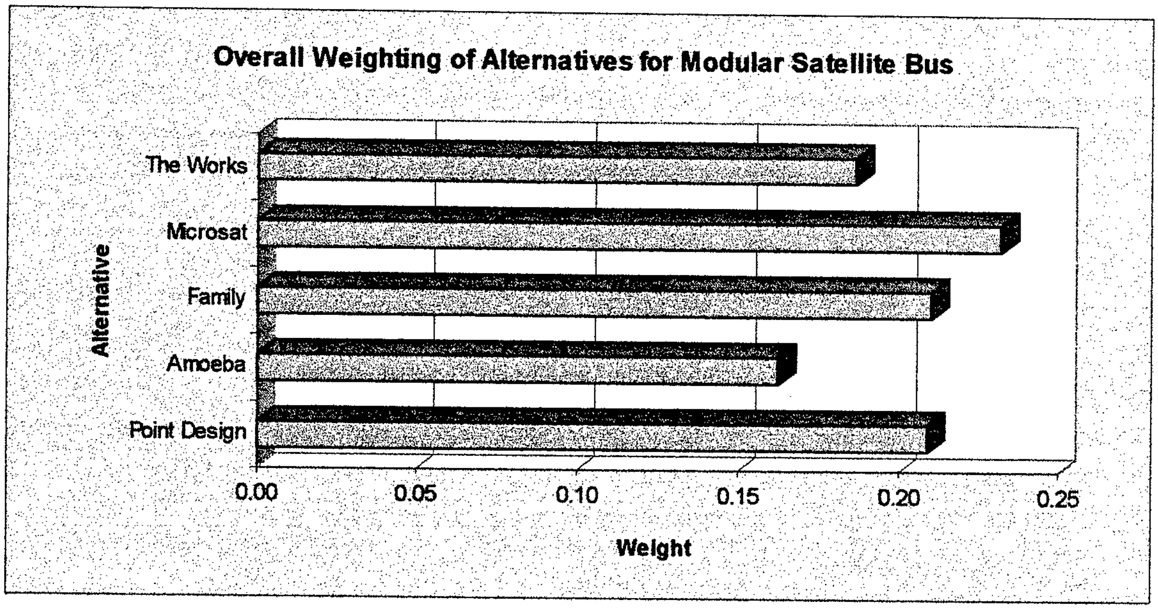


Figure 9-4: Scenario Results for *Acquisition Cost* as Most Important for Fleet Cost

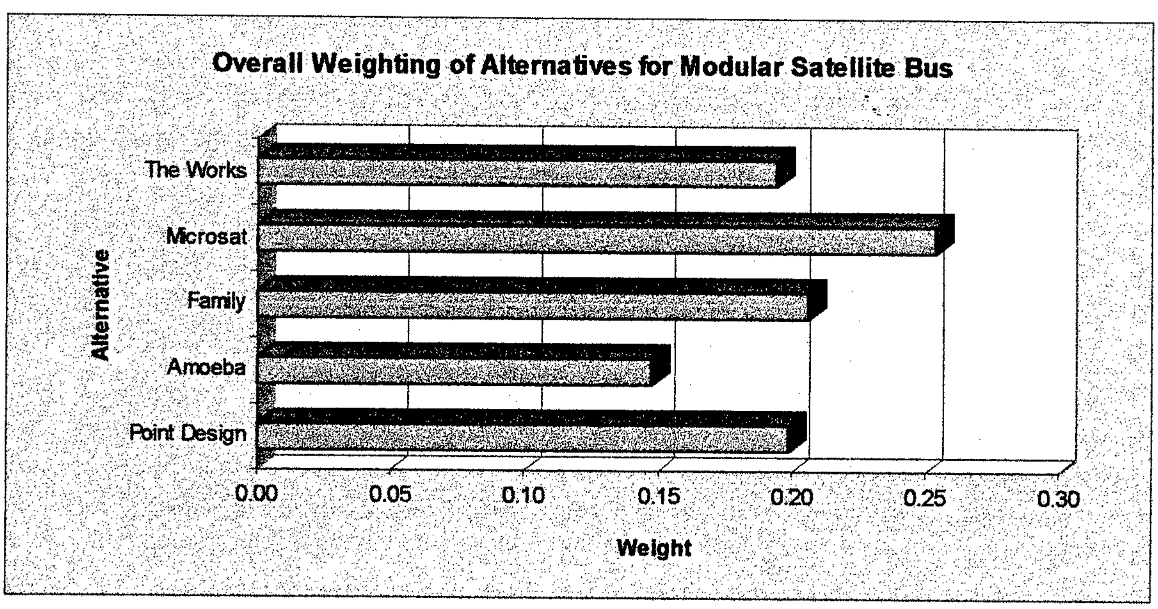


Figure 9-5: Scenario Results for *Operation Cost* as Most Important for Fleet Cost

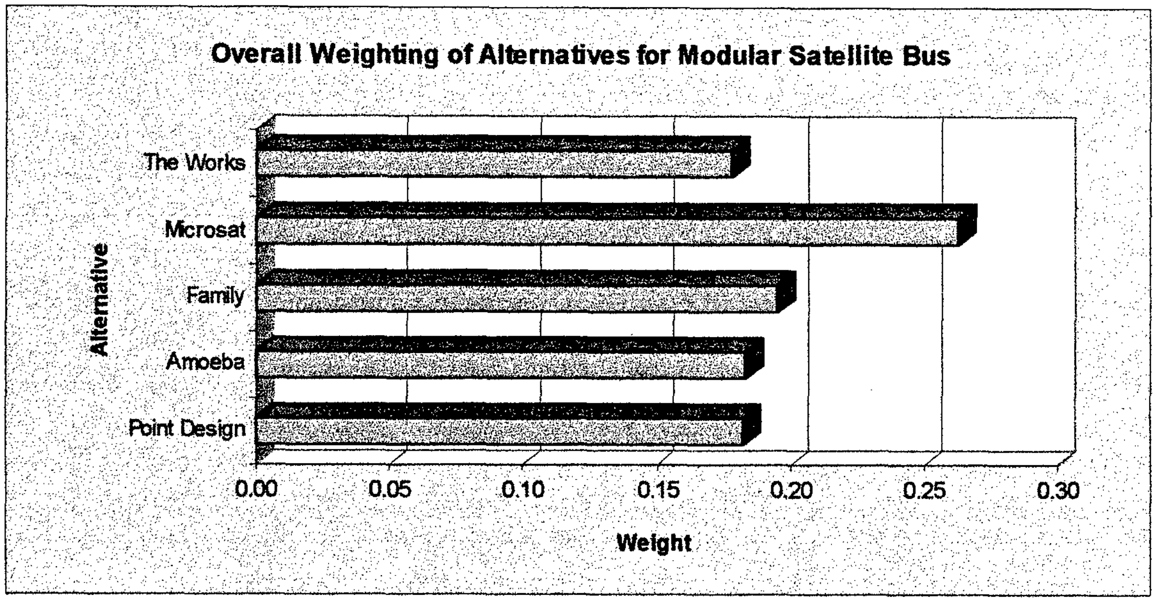


Figure 9-6: Scenario Results for *Retirement Cost* as Most Important for Fleet Cost

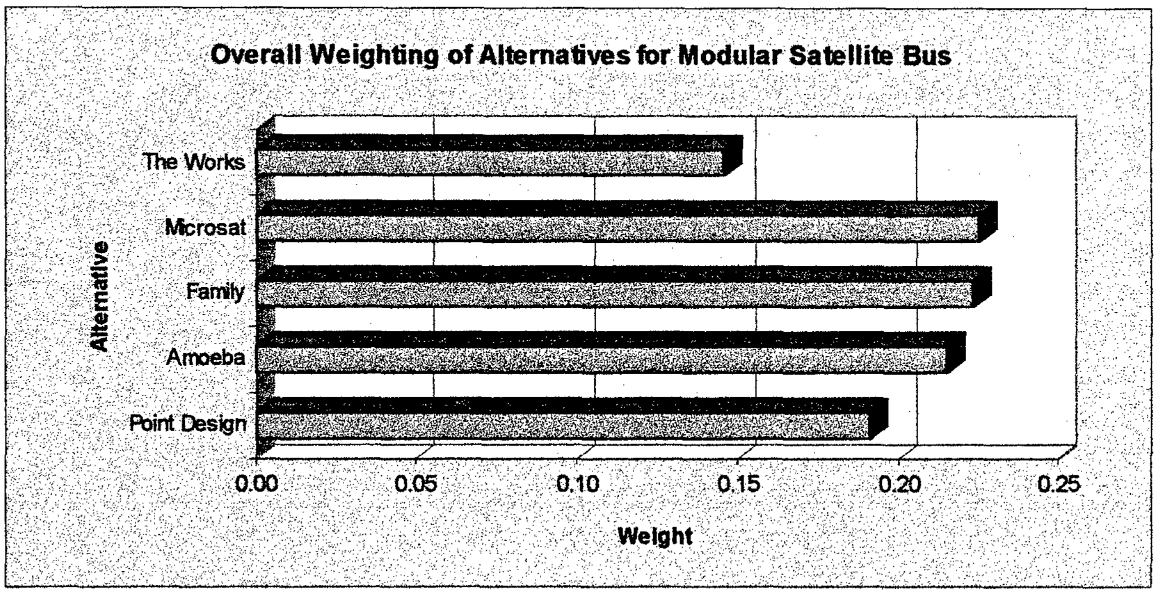


Figure 9-7: Scenario Results for *Flexibility* as Most Important for Effectiveness

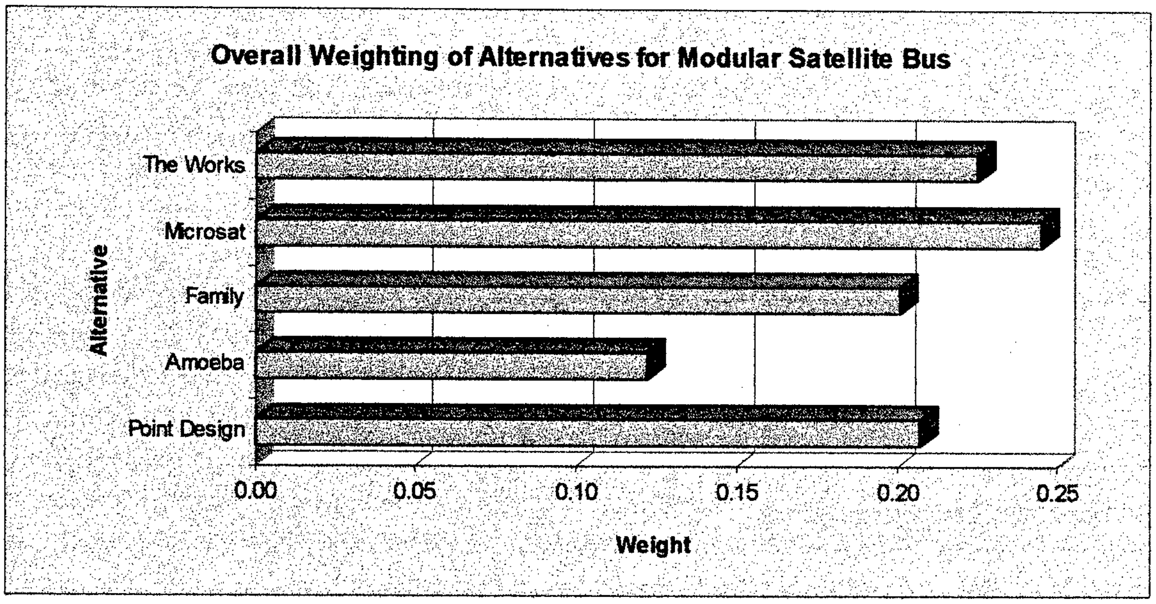


Figure 9-8: Scenario Results for *Availability* as Most Important for Effectiveness

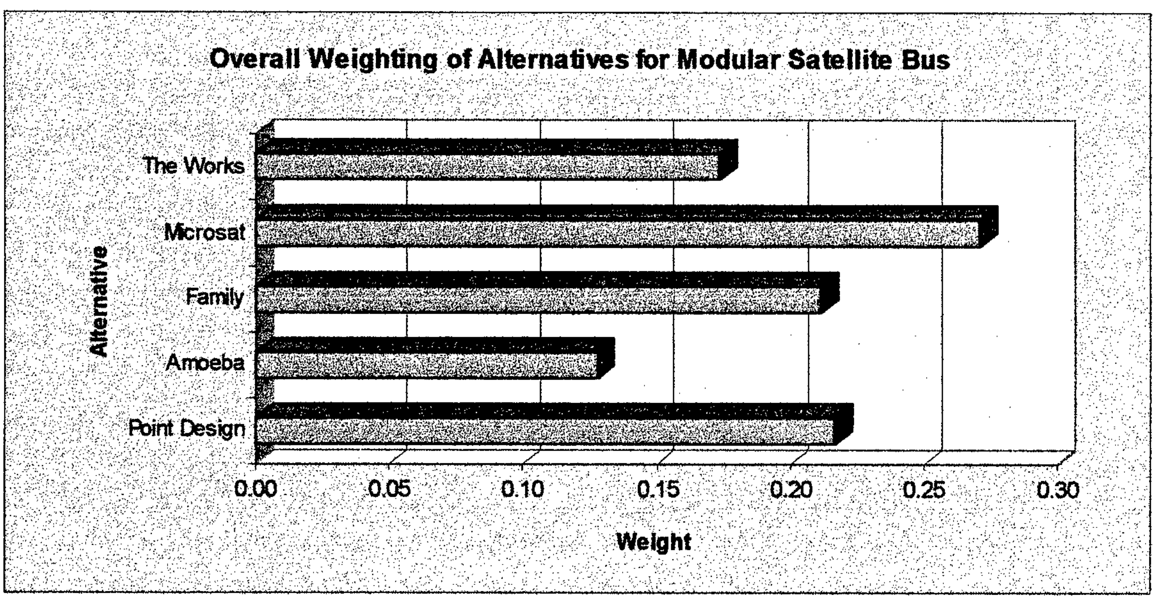


Figure 9-9: Scenario Results for *Responsiveness* as Most Important for Effectiveness

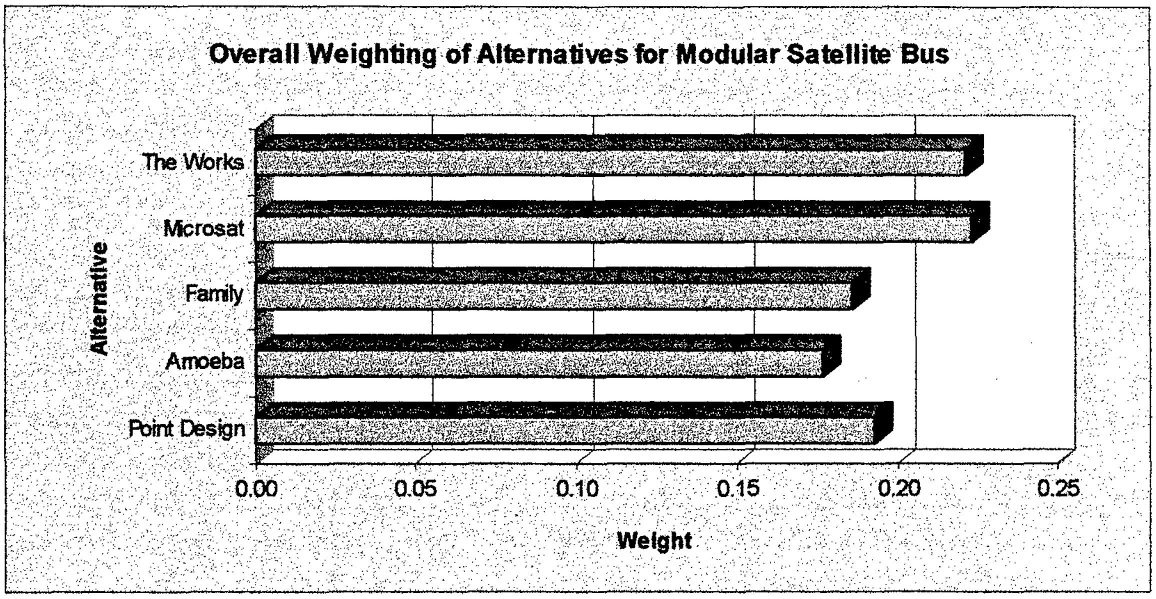


Figure 9-10: Scenario Results for *Capability* as Most Important for Effectiveness

Overall, the System Evaluation step determined the Microsat architecture to be the best solution to all scenarios. However, the Microsat option does not always lead the other architectures by a wide margin, and further investigation may reveal another solution to be the best option overall.

10. Decision Making

10.1 Introduction

In this section, the consequences of the alternatives which are developed in system analyses relative to the objectives will be evaluated and these evaluations will be incorporated into the decision criterion so that all alternatives can be compared relative to the criterion. This will enable one or more alternatives to be selected for advancing to the next iteration. Also this section will include communicating the results of our process to this point, scheduling subsequent efforts, and assigning priorities for subsequent action.

10.2 Results

The Results of the System Engineering Group's analysis of pair-wise comparison showed that The Microsat alternative ended up with the highest utility score.

In the minimizing acquisition cost, point design had higher weighting because of ease of manufacturing, adaptability to the off the shelf technology wherever possible, low cost in testing and launch integration, and being less complex. On the other hand Amoebae and Works had less weighting since they had more complexity and integrity in manufacturing and testing.

In the minimizing operation cost, the Microsat had higher weighting because in the tactical scenario it can be operated by individual mobile units and it may not need complex and integrated control units compared to the works. Testing, launch integration, survivability and reliability decreased the operational costs. On the other hand integrational complexity increased the difficulty in system and user sides in terms of standardization, subsequently resulted in higher operational cost.

In the minimizing retirement cost, ease of de-orbiting and ending the mission life were two important factors. Under these conditions Microsat obviously had higher weighting and Works had lower weighting.

For flexibility, Amoebae provided features of modular flexibility that allowed it to be adopted to a wide range of missions and payload configurations. Also it could easily satisfy multiple missions simple by changing sensors and software. The Works' long life limited it to adopt the newly arising changes in the space environment. On the other hand, specialization to specific missions were important aspects for the other alternatives.

For availability, the Works' multiple payload configuration enabled it to assign different missions. Also because of long mission lifetime, it has to be more reliable for on-board data processing, uplink and downlink communications systems. Amoebae's complexity in modular integration decreased its reliability.

For responsiveness, miniaturized standard design of satellites and payloads (mass-produced and deployable in dozens of units) could offer responsiveness to newly arising situations in tactical environment. Thus Microsat had higher weighting. But in terms of standardization Amoebae had more disadvantages.

For capability, the Works' multiple payload configuration, on-board data processing, and operational autonomy made it more capable for different mission requirements at the same time. The other three, Family, Microsat, Point design, had lower weighting in that sense.

10.3 Sensitivity Analysis

According to scenarios generated in system evaluation part seven sub-scenarios are summarized in the Table 10-1.

Table 10-1: Scenario Summary

Fleet Cost	1. Microsat	2. Point Design	3. Family	4. The Works	5. Amoebae
			Ranking		
Most Important	1	2	3	4	5
Acquisition Cost	Microsat	Works	Family	Point	Amoebae
Operation Cost	Microsat	Point	Family	Works	Amoebae
Retirement Cost	Microsat	Family	Point	Amoebae	Works

Effectiveness	1. Microsat	2. The Works	3. Family	4. Point Design	5. Amoebae
			Ranking		
Most Important	1	2	3	4	5
Flexibility	Microsat	Family	Amoebae	Point	Works
Availability	Microsat	Works	Point	Family	Amoebae
Responsiveness	Microsat	Point	Family	Works	Amoebae
Capability	Microsat	Works	Point	Family	Amoebae

11. Recommendations

11.1 Recommendations for Continued Investigation

Initial investigations into small tactical satellite design and evaluation focused efforts (necessarily) on high-level design architectures. This high-level approach accomplished the goals of ensuring the familiarity of Team members with the systems approach to design; introducing less space-savvy members to satellite design, production, and operation; serving as an initial "design compass" to focus not on specific architectures, but on those aspects of the value system which were most important; and to provide insight into possible implementation schemes for the final design combined with an overall design architecture and concept of operations (CONOPS).

After evaluation of the large-scale satellite concept architectures, the system design process must shift its focus to specific satellite design exploration and the incorporation of those aspects of the value system which provide the greatest payoff (as concerns the satellite design). This process will involve research into the specific subsystems and specific components of those subsystems and their impact on the overall satellite as a whole.

Further refinement of the problem definition and value system will be facilitated by further contact with the sponsor, space and industry experts, and research into the individual subsystems.

System modeling and evaluation will incorporate a computer-based (MATLAB) design tool for both designing and evaluating candidate designs. A computer-based modeling scheme will facilitate the convergence of candidate designs into cohesive systems, as opposed to collections of components. This approach is intended to converge on potential solutions, based on Value System Design. Current architectures will be used as a starting point, but new concepts can be generated.

The concept of a Microsat certainly lends itself to small launch vehicles, but it does not necessarily support the range of payloads as put forth by the sponsor. Power requirement is a big variable as relates to a potential range of payloads. The task of attitude determination and control is no small chore either, since varying payloads with their necessary power support can significantly alter dynamics from one configuration to the next. It seems that either a satellite bus must be overpowered and possess over capable dynamic control for most mission modules (The Works), or these factors must be alterables in the form of interchangeable components (similar to Amoebae) or as two or more basic fixed designs (Family). All of these concepts are similar, though, since some satellite bus is on hand with the payload required to fit within specifications provided by the bus, not the other way around.

PHASE II

12. Introduction

Subsequent iterations of the systematic approach lead to revisions, clarifications, and refinements to both the process and the product. The team learned that the original systematic approach used in Phase I was not effective in narrowing the scope of the problem and developing design alternatives. This required the team to create a new process. This section documents the team's work in applying the innovative approach to the design of a generic, small standardized satellite bus for tactical applications.

Phase II follows the outline of the newly created systematic approach and discusses the changes and refinements made since the work performed in Phase I. It is assumed that the reader has a working knowledge of the new design process being applied and is cognizant of the work performed in the first iteration. Specifically, background data and definitions of the process are not included in this section.

This section documents the design team's understanding and scope of the problem as refined by the second iteration of the systematic approach. In addition to providing the details on the refined problem statement and subsequent changes to the value system design, a new section was added to discuss the trade studies that were performed on both system and subsystem levels. The section also provides the details on the efforts made in the creating an integrated model and the alternatives that were created. The section concludes with an in-depth analysis of the design alternative and suggests methods for implementing the designs.

13. Problem Definition

Gaining insight into satellite design and probing the different aspects of the proposed problem were the main focus of the first iteration. In the second iteration, the team focused its effort on studying and understanding the functions of the satellite subsystems and examining the factors that influence the satellite bus design. This led to a more detailed examination of the tasking statement and the factors that influence the problem. The first item re-examined was the problem statement.

The team examined the problem statement and decided that the word 'modular' was not required to be part of the statement. The team decided that being modular was more of a design option rather than a goal of designing a generic satellite bus. The team felt that by including 'modular' in the problem statement, the number of possible design alternatives would be severely limited. The word was removed and the refined statement is provided below. The new statement was referred to throughout the design process. This ensured that the team's efforts remained focused.

13.1 Problem Statement

The refined problem statement reads:

Design a rapidly deployable, tactically oriented, satellite bus to enhance theater operations. This satellite bus is to support missions in the Pegasus and Lockheed-Martin Launch Vehicle (LMLV) weight class.

13.2 System Boundary

Since the system boundary defined the elements of the problem that could be controlled or manipulated, the team decided that the best way to narrow the focus and scope of the study was to refine this area. The boundaries were divided into two distinct environments; an external environment and an internal environment. Items that existed in the external environment influenced the solution space for the problem, but were not items that the design team could control. These items were considered outside the team's scope with respect to redesigning components or changing concepts of operation. Items contained within the internal environment were aspects that could be controlled by the design team and were subject to trade studies. Figure 13-1 provides a graphical representation of the problem's system boundaries.

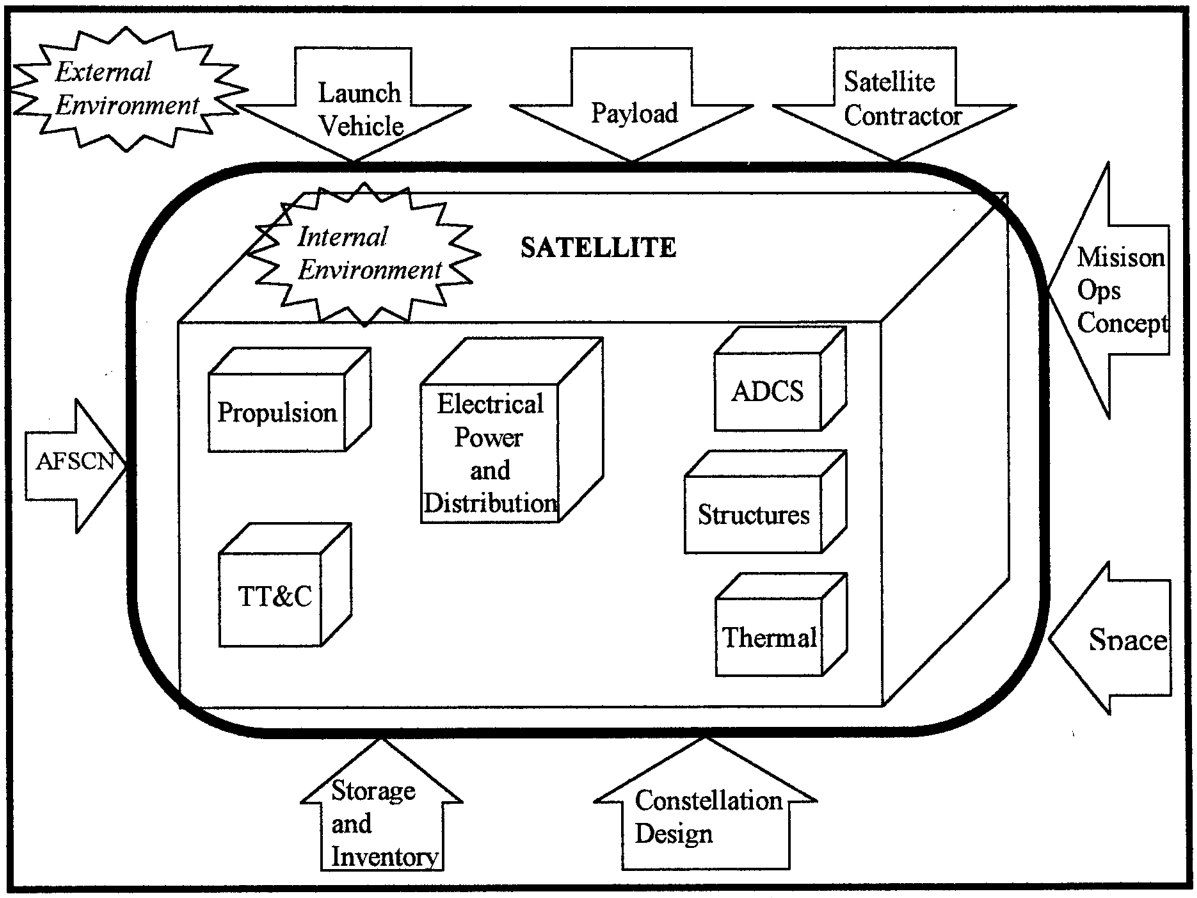


Figure 13-1: System Boundaries

The external boundary was comprised of the following items: the launch vehicle, the payload, the satellite contractor, the mission operations concept, space, the constellation design, the storage and inventory concept, and the Air Force Satellite Control Network. Aspects of each element are described below.

- The launch vehicle: The design team examined the system requirements for placing the vehicle into orbit. The team decided to use the Pegasus XL launch vehicle for this design study. Aspects of the launch vehicle that had an influence on the satellite bus design were launch preparation time, mass-to-orbit performance, satellite-to-launch vehicle integration constraints, and fairing constraints. Launch vehicle development and integration of launch vehicle stages were outside the realm of the design team's control and were not subjected to trades or redesign.

- The mission module: A number of different mission modules were examined. These included remote sensing payloads such as multispectral imaging (MSI) systems, synthetic aperture radar (SAR) systems, infrared (IR) systems, and laser designators. Specific mission modules were not designed by the team to be integrated onto the generic bus. The bus design was influenced by the mission module's data storage/health and status requirements, thermal loading, power requirements, mission requirements, and required pointing accuracy's.
- The satellite contractor: A satellite company has a definite influence on the design when it comes to manufacturing the actual satellite bus. Additionally, different companies have different manufacturing processes. Due to the preliminary nature of the design study, the team considered items that would create manufacturing difficulties, but did not perform extensive research into actual satellite manufacturing.
- The mission operations concept: The methods the United States Air Force (USAF) employs to perform its satellite missions had an influence on the satellite design. The design team did not attempt to change or modify the way the USAF does business. However, understanding the constraints and requirements needed to perform satellite operations was a prerequisite for this design effort.
- Space: This was the actual operating environment in which the satellite bus would perform its mission. It was important that the design team understood what effects

space could have upon the satellite bus. The team had to design the vehicle in such a way that it accounted for the space environment.

- The constellation design: The actual use of the vehicle and constellation deployment was left to the satellite user. The orbit choice and number of satellites to be placed into orbit will be contingent upon the user's need. The team provided a satellite bus design that attempted to maximize its utility to the user.
- The storage and inventory concept: It was taken as an assumption that the satellite and its components would be stored and maintained in an appropriate clean room environment while the satellite was on the ground. Therefore, the team did not design any aspects of the storage and inventory process. However, consideration was given to this storage and inventory process when design alternatives were developed.
- The Air Force Satellite Control Network (AFSCN): Since compatibility with the AFSCN was required by the decision maker, aspects associated with this satellite control network had a major influence on the design. Satellite components such as receivers and transmitters had to operate at Space Ground Link System (SGLS) frequencies. The team made no attempts to redesign any aspect of the AFSCN.

Items contained within the internal environment included the satellite bus subsystems, the launch vehicle interface, and the mission module interface. Operational concepts directly related to the use of the bus were considered within the boundary of the system. The team had the freedom to modify concepts such as on-orbit command and control, mission module and launch integration, and sensor data processing. The team decided that the concept of having two different on-orbit command and control systems to the satellite was well within the scope of the bus design.

13.3 Needs

Further revisions were also performed in the needs area. This was accomplished because the team wanted a clearer understanding of the problem. The categorized lists created in the first iteration only provided generalized groupings of the needs. A deeper understanding of the problem required more definition be given to the problems needs. It was believed that if a needs definition could be tied to the system's boundaries or the decision maker's views (Rooney, 1996), it would offer a better understanding of the problem. The refined needs definitions are provided below and incorporate the sponsor's views or the system's boundary considerations as appropriate.

- Mass: The satellite design had to be optimized to support as many mission module types as possible. Mission modules with masses between 23 and 114 kilograms had to be supported and fit within the constraints of a Pegasus XL launch vehicle. It was thought that better designs would provide more mass and volume to the mission module yet still supply ample power and interfaces.
- Responsiveness: Possible design alternatives had to consider the rapid launch of a satellite constellation. The bus/mission module combination had to be easily integrated to meet the need for rapid deployability and tactical applications.
- Sensors/mission modules: Different mission requirements were considered; i.e., electro-optical (EO), infra-red (IR), laser designators.

- Pointing accuracy: The question of pointing accuracy had to be considered when trying to achieve 1 meter resolution during an imagery pass of 5-10 minutes per orbit.
- Power: The satellite design had to support peak power requirements up to 1 kilowatts and average power requirements of 300-500 watts.
- Orbital maintenance: The satellite had to operate at a minimal orbital altitude of 300 kilometers for a mean mission duration of 12 months.
- Telemetry, Tracking, and Command: Satellite design alternatives had to be compatible with the AFSCN. Data downlinks had to support near real-time transmission of 1 meter resolution imagery data. Encryption and deception provisions were also required.
- Data storage: Designs had to support on-board storage of up to 100 images.
- Data Processing: Design alternatives had to support minimal on-board processing and data compression algorithms for transmission of images to ground station.

13.4 Mission Module Overview

The problem involved with the design of a "generic" satellite bus is that, because the bus is to be generic, it cannot be designed to one specific payload or mission. It must support the requirements demanded by all foreseeable missions within the scope of the overall design.

13.4.1 Background and Scope

The payload or mission equipment of any spacecraft is generally considered to be that particular spacecraft's reason for existing. The payload is, after all, comprised of the equipment which the spacecraft owners and users desire to employ (from the space vantage) for the collection or distribution of very specific mission information. Consequently, satellite designs in the past have always focused on this specialized equipment, functionally separating it from the rest of the vehicle (satellite bus). Within this paradigm, payloads tended to be as large, expensive, and/or as powerful or capable as possible. The bus was basically designed and built to support that particular payload (i.e., the bus was built up "around" the payload). Thus, because of the specific nature of a spacecraft's payload equipment, as well as owing to the fact that all satellites are basically manufactured by hand, individual satellites tended (and continue) to be unique. Similarities in design and equipment among satellites of the same constellation or "family" are more numerous, but even these satellites have been and continue to be dissimilar in some areas, due to the addition of features, change of specifications, or flight experience from earlier designs. All of these factors, in addition to the slow historical launch rate, tended to drive up costs. A "vicious circle" of spiraling costs ensued and continues today, driving designers to build fewer, more reliable, more capable, and larger satellites. The larger vehicles compounded the launch availability problem due to the fact that larger boosters became necessary for the larger vehicles -- larger boosters take longer and are more costly to integrate.

Shifting equipment and design focus AWAY from the payload components forces a shift in the spacecraft design paradigm. It focuses on the vehicle itself, the bus, as a starting point for employment of special sensors or other equipment from space. In this paradigm, the payload simply becomes yet another "component" which must be integrated into the vehicle as a whole -- the payload specializes or tailors a standard vehicle to a specific mission or purpose. This paradigm is analogous to a multi-role fighter aircraft being outfitted with a particular weapons load for the performance of a specific mission. The fighter is the "standard vehicle" which can then be used for a variety of missions. This particular paradigm requires the payload designer to produce payload equipment packages which can seamlessly interface with and "take a ride" on a satellite bus which has already been designed (or built) and can provide all the payload support functions necessary.

This intimate interface between the payload and the spacecraft bus then becomes the focus of design efforts. The payload equipment package (from here on "mission module") must conform to a standard set by the bus design; however, the bus must be able to support the proper functioning of the mission module by providing (as required by specific mission type) power, data and command handling, thermal protection and/or isolation, stability and pointing accuracy, structural and mechanical integrity, slew rate/accuracy, and, of course proper orbital position over the surface of the Earth. Therefore, the design problem further distills to the generation of a "starting point" or baseline set of mission module support requirements from which an iterative process of bus-design-and-mission-module-interfacing may commence. The most desirable conclusion from this process is a bus design that not only meets a set of given mission module requirements, but also supports a CONTINUUM of mission module requirements (i.e., non-discrete requirements values), while providing a set of standard volumetric, mechanical, data, power, and thermal interfaces to which mission modules may be physically designed.

But where to start? The initial starting point for an eventual (desired) continuum of missions must start with a standard set of requirements which fully span the "requirements space". That is, a good initial set of requirements should span the full range of foreseeable requirements which the bus design must support. It follows that in order to derive an initial set of requirements, an initial set of missions must provide sufficiency in spanning the requirements space. These mission areas must: 1) define the initial points from which requirements estimation relationships (RERs) may evolve, and 2) be sufficiently "expandable", thereby providing the desired continuum or range of requirements over which the bus architecture and design iterate and converge.

This problem is well suited to computer model support because of the iterative design approach which this problem requires. The course of several iterations of mission module requirements generation combined with bus design attempts (analogous to a Newtonian or similar approach to numerical estimation) will converge a solution set of bus designs as candidates for further (operational effectiveness) evaluation. The issue, however, returns to the fact that RERs must provide the requirements inputs to the model and the initial set of missions must sufficiently characterize the space which the RERs must span.

13.4.2 Mission Module Support Requirements

The basic space mission type is an Earth observing or remote sensing mission; however, this simple mission type is further subdivided into a plethora of specific sensing missions directly related to military operations and directly applicable to a tactical (in-theatre) environment. Furthermore, within this general mission area, one finds a wide range of payload equipment and sensors, varying in mass from a few to several hundred kilograms and varying in power requirements from a few to several hundred kilowatts.

These characteristics make this space mission area advantageous to the task of generating initial mission requirements.

The next step in defining initial mission module requirements is to generate some discrete requirements values for the specific mission types. The specific mission module characteristics/requirements under consideration and of concern to the mission module designer may include:

- electrical power
- vibrational stress
- thermal stress
- thermal isolation
- structural integrity
- mechanical interfaces
- electrical interfaces
- telemetry formatting
- equipment volume
- equipment mass
- data throughput
- data storage
- data down-link
- platform slew rate
- platform pointing
- platform stability
- positional (surface) revisit
- positional (orbit) tolerance

For purposes of this design study, the grouping of requirements into larger "requirements measures" which capture the meaningfulness of the "bit level" parameters facilitates generation of RERs which are easily modeled. Additionally, a design study involving these lower-level requirements is beyond the scope of this investigation. Therefore, it is advantageous to attempt to either capture these requirements within higher-level expressions or treat them as ancillary information along with other "payload designer-generated" information such as mission module costs. Other requirements information may be treated as "built into" the design, such as vibrational stress and mechanical/structural interfaces/integrity. Thus, a "higher-level" list of characteristics/requirements suitable for concept exploration may include:

- available power
- available launch mass
- available launch volume
- pointing/stability requirements
- data handling requirements
- data storage requirements
- mission data down-link support
- thermal protection and/or isolation

The aspects of the satellite-to-mission module interface which will be most important to the mission module designer will be mass, volume, power, and data storage budgets available for the mission module. These budgets will provide the mission module designer with limits within which the mission-specific equipment must operate.

Similarly, the most important considerations for the bus design will be the support of those baseline power, mass, stability, pointing, data handling, data storage, and thermal isolation requirements necessary to accommodate all of the baseline mission module types. These types include applications spanning basic electro-optical radiometers, multispectral imagers, LASER/LIDAR systems, and synthetic aperture RADARs. These mission module types were chosen for their diversity and their applicability to tactical space applications. Because of the generic nature of this study, and due to the fact that specifications for military systems (within these categories) are either classified or unavailable at this time, estimates for mass, power, volume, and other specific requirements had to be generated from experience, remote sensing class notes, SMAD, and the few analogous commercial, scientific, and civilian applications available for inclusion. For purposes of this design study, however, which focuses on the design of a specific satellite bus (not the mission modules), lack of specificity of mission module designs will not impact the overall design of the bus. The purpose of a discussion of mission module requirements will provide, in many cases, valuable performance requirements to be met by the specific subsystems of the satellite bus (e.g., pointing accuracy requirements will drive decisions made about attitude control system components). In all cases, extrapolation of estimates was conservatively overestimated in order to provide sufficient design margins.

13.4.3 Specific Mission Module Types

13.4.3.1 Electro-Optical Imaging (EO)

The least expensive, lightest weight, lowest power, and probably the widest used payload type for tactical missions is the simple yet capable, high-resolution camera system. Systems of this type gather electromagnetic (EM) radiation in the visible (VIS: ~0.37-0.75 micron) and sometimes ultraviolet (UV: ~0.15-0.39 micron) regions of the EM spectrum. The military utility of this type of imagery dates back to the first days of placing observers in balloons, and later placement of cameras in reconnaissance aircraft. One disadvantage of this mission module type is that, because it depends upon the reflected illumination from the Sun, a satellite equipped with this type is most effective in Sun-synchronous orbits (see orbit tradeoffs discussion).

The configuration for this basic mission module type consists of a telescope housing the optics, incorporating a compact "folded" optical design, and a sensor suite housing the detectors (a charge-coupled device, or "CCD" array). The chosen orientation for this mission module and others is axial, nadir-pointing from the "top" shelf of the spacecraft bus orientation. Estimations are based conservatively (i.e., overestimating to provide a design margin) on and extrapolated from the optics package and cameras used on the (1994) Clementine 1 mission and proposed for the (1998) Clementine 2 spacecraft. These high resolution cameras were originally developed for use on board small satellites deployed at LEO and are, therefore, appropriate to this design study. Estimations use aperture size (in centimeters) as the basis for all of these first-order estimates of mass and power. These estimates will vary, of course, depending on individual detector size (anywhere from 5-20 microns) and efficiency of optics design/construction. Overall field of view (FOV) for these systems will be totally driven by the size of the CCD array. For example, a system with an instantaneous field of view (IFOV -- the FOV for one detector) of 2.86 microradians (one-meter nadir spatial ground resolution from a 350 km orbit), will require a 1000 by 1000 detector array to cover a 1km by 1km ground target. Note that a payload designer, depending on the performance capabilities desired, determines the specific volumetric/mass characteristics for a mission module as well as the particulars of the arrangement of the optics; therefore, estimation relationships are intended to be initial "first order starting points" if no specific design is determined ahead of time.

Table 13-2 summarizes some estimated EO mission modules and their characteristics. Optical diffraction limits use the radiometric resolution equation

$$\phi = 1.22\lambda/\mathbf{D} \qquad \textbf{(Eqn 13-1)}$$

where ϕ is the IFOV (in radians), λ is the wavelength (in meters) of the EM radiation gathered by the optic, and D is the aperture (in meters) of the optic. For the EO mission module, a central wavelength of 0.5 microns is assumed.

Table 13-2: Electro-optical (EO) Mission Module Estimations

Aperture (cm)	Diffraction Resolution (μrad/m)	Mass (kg)	Volume (m^3)	Power (w)
30	2.03/0.71	23	0.035	7.5
40	1.53/0.53	41	0.082	10.0
50	1.22/0.43	64	0.158	12.5

13.4.3.2 Multispectral Imaging (MSI)

Advances in image processing as well as improvements in detector performance over the past few years have made MSI a high-demand payload. The MSI mission module uses several different arrays of detectors (CCD arrays), each optimized to detect a specific band of EM radiation. Image processing produces simultaneous images of a target area characterized at various regions of the EM spectrum. These regions may include spectral bands from UV, visible, NIR (near infrared: ~0.8-3.0 microns), MIR (middle infrared: ~3.0-6.0 microns), and LWIR (long-wave (thermal and extreme) infrared: ~6.0-30.0 microns). Intensity levels at specific wavelengths may indicate, through analysis, a particular activity, characteristic, or object within the field of view. By overlaying and comparing the levels of intensity at specific wavelengths from a single target, many characteristics of the target and subsequent target identification may be determined by comparing the received spectra to known spectra (predetermined spectra for specific substances -- a particular type of vehicle paint, for instance). Thus, MSI image processing and analysis is related and/or analogous to spectroscopy, in which the characteristic elements in a compound may be discerned by comparing (with established spectra for suspected elements) the spectra returned from a (bombarded) sample. Though not as dependent upon illumination from the Sun for radiometric sensing (infrared sensing, especially thermal imaging, operates effectively on the night side of the Earth), the MSI mission module is most effective in a Sun-synchronous orbit, due to its normal inclusion of UV and visible region detectors. These orbit considerations will vary, in accordance with varying detector types and specific mission objectives.

The key in MSI processing is the "suspected" ingredient -- spotting the component spectra which best "match" those returned from the target (this is basically a "pattern recognition" problem for image processing). The utility to military planners, of course, will be the ability to ignore traditional camouflaging techniques which are necessarily designed to subvert visible identification (i.e., MSI tactical users can discriminate between a tank, a tank under camouflage, and a decoy tank under camouflage -- all due to subtle differences in spectral characteristics). Development of decoy and/or camouflage techniques to defeat this detection scheme will prove much more difficult and technically demanding than traditional visible camouflage techniques due to the sophistication of the methods involved. Another high utility function of the MSI payload is the capability to determine minute changes in a target, based on changes in the spectral characteristics of the target. This situation is analogous to taking a "before" and "after" picture of an object to determine changes in position, motion, heating, cooling, loading, shape, or orientation.

A primary performance characteristic for these systems is "spectral resolution" which gives a measure of the bandwidth of one spectral sample (pixel) from the mission equipment. This is analogous to the traditional spatial resolution which gives a measure (along the surface of the Earth) of the width of one image sample (pixel). As the spectral resolution of the system improves, the number of spectral channels ("bins") resolvable across one pixel increases. A high degree of spectral resolution provides a very accurate spectral mapping of the target area. What image processing experts have found is that, given a known spectral signature for an object and sufficient spectral resolution (channels), it takes significantly fewer pixels to spectrally identify that object than it would take to spot the same target using a (finer) spatially-resolved EO system.

MSI payloads can use the same type of optical design as do EO payloads. The difference between the two mission module types resides in the detector types and support for those detectors, as well as the wavelength of EM radiation to which the optics and detectors have been optimized. Detectors that share optics normally share optics which have been simply optimized to the central wavelength of the full range of the detectors. Most MSI payloads include a visible EM detector to include with the other detected wavelengths. Basically, the MSI payload splits the target irradiance (received by the optics) into several beams, routing specific bandwidths to their appropriate detectors. The routing method may take any number of different forms, including diffraction gratings (this tends to be the most efficient method), prisms, special fiber optic cables, or elaborate reflection/transmission optics. Additionally, for thermal imaging detectors (LWIR) a varying requirement for cryogenic cooling will be required, depending on chosen detector type, semiconductor material, and desired sensitivity.

Physical characteristics for the MSI mission module are similar, but more massive and more power-hungry, than the EO mission module, due to the additional support required for the different spectral detection bands. Estimations are extrapolated from MSI payloads based on the Miniature Sensor Technology Integration-3 (MSTI-3) program and the MSI sensor suites for Clementine missions 1 and 2. As with the EO payloads, the will be overestimated when compared to these specific examples -- again, accounting for a design margin. MSI payload characteristics will vary according to specific design and performance requirements, and the estimations account for this likelihood by allowing initial, first-order estimates on mass, power, and cost, to be edited.

Estimates for MSI mission modules and their vital characteristics are included in Table 13-3.

Table 13-3: Multispectral Imaging (MSI) Mission Module Estimations

Aperture (cm)	NIR (1.5μm) Diffraction Resolution (μrad/m)	MIR (4.0μm) Diffraction Resolution (μrad/m)	LWIR (10.0μm) Diffraction Resolution (μrad/m)	Mass (kg)	Volume (m^3)	Power (w)
30	6.1/2.135	16.3/5.69	40.7/14.23	28.5	0.039	60.0
40	3.75/1.3125	12.2/4.27	30.5/10.68	50.3	0.089	80.0
50	3.0/1.05	9.76/3.42	24.4/8.54	78.7	0.169	100

13.4.3.3 LASER/LIDAR Applications

Using optics similar to the EO package -- in some "functionally dense" mission modules, the VERY same optics as the visible camera/detector -- the LASER imaging payload adds a LASER head and power supply (LASER pump) in order to illuminate a target with a specific wavelength of EM radiation. This method effectively increases the irradiance (the number of photons) from the target that is returned to the detector, thereby enhancing imaging performance for that given wavelength. Concurrently, due to the coherence of the returned signals, highly accurate ranging and range rate (velocity) determination is possible (similar to a RADAR). Because the LASER mission module provides its own illumination, it could operate effectively in orbits other than Sun-synchronous and at night, making it more tactically available. Due to the narrow beam, however, LIDARs are not normally used to illuminate or track moving targets. These mission modules can produce very accurate three-dimensional, narrow-swath imagery (especially for range determination) , making them well-suited for topographical missions and atmospheric/meteorological (cloud system) observations. Of course, an active sensor system such as this will require much more power than passive systems.

A major drawback for the LASER or LIDAR system is atmospheric attenuation of the LASER light illumination from the spacecraft. Atmospheric attenuation varies with varying wavelength of light from the LASER. Any mission designed to perform active sensing of targets on or near the Earth's surface will require more and more power pumped into the LASER, depending on the desired level of irradiance incident on the target and the wavelength of the LASER. Power requirements increase substantially if the system must perform its mission through atmospheric disturbances, such as storm clouds over the target. This increasing power requirement will drive up power system mass and overall spacecraft mass and cost. LASER mission estimators are based on the LASER head and power supply flown on Clementine 1 and proposed for the Clementine 2 spacecraft, due to its initial development for LEO applications (these LASERs are in the 100-200 watt range and weigh a few kilograms) (BMDO, 1994; Clementine Team, 1996). These estimators provide conservative initial values for mass, size, and power, which may then be updated if more accurate specifications are known.

A good example of a surface-sensing topographical/meteorological mission payload is a proposal for a NASA spacecraft for launch around 2000 (as a technology demonstration). This system incorporates a 500-watt LASER and will achieve an estimated altitude accuracy (of surface features) of one to two centimeters (NASA, 1995).

LASER technologies also may extend into more exotic applications, such as extremely high data rate communications packages and tactically-applied LASER designation systems (see Future Technologies and Continuing Investigations). While these experimental and/or developmental applications are not specifically addressed in this design study, a theoretical mission module for a more exotic LASER application may still be modeled by simply editing the pre-generated estimates for mass, power, and other characteristics. Additional mission module components may also be modeled by simply specifying the new component characteristics and adding them to the mission module. Table 13-4 summarizes some estimated LASER/LIDAR mission modules and their characteristics.

Table 13-4: LASER/LIDAR Mission Module Estimations

Aperture (m)	LASER Power (w)	Mass (kg)	Volume (m^3)	Power (w)
30	300	38.7	0.044	318
40	250	56.7	0.089	274

13.4.3.4 Synthetic Aperture Radar (SAR)

By far the payload with possibly the greatest potential tactical "payoff" is the Synthetic Aperture RADAR or SAR mission module, which can produce very high resolution images through intensive image processing. SAR mission modules, like

LASER-based mission modules, are active sensing systems and, as such, generally require an order of magnitude greater power to operate than passive systems (EO and MSI).

An advantage of the SAR mission module is that, because of the microwave frequencies used (typically 1-25 centimeters), its signal is much less susceptible to atmospheric attenuation. SAR sensing capabilities are applicable in all atmospheric conditions, regardless of weather over the target. Additionally, SAR operates effectively in either sunlit or night conditions. SAR also has the capability of imaging underground, enclosed, or physically covered objects by penetrating the enclosures or ground covers. This day-night, all-weather capability makes SAR a highly desirable mission module type for tactical space applications. Tailoring of many SAR design characteristics, including power level, operational wavelength, physical aperture size, antenna gain, and pulse repetition frequency (the rate at which the SAR alternates between transmitting and listening) correspondingly tailors the equipment to the user's requirements for imaging at different resolution settings as well as imaging different surfaces, materials, or features (Brodsky, 1992: pp.271-274). If the antenna for the SAR is steerable, either physically (gimballing mechanisms) or electronically (phased array architecture), and the satellite bus can support accurate pointing capability (typically within 0.1 degrees accuracy, with 0.05 degrees attitude knowledge), the SAR may perform "spotlighting." Spotlight mode for a SAR trains the transmitted beam of microwave radiation at a target for a longer period of time, mimicking a "staring" system and producing image spatial resolutions generally two times to four times better than a SAR mission module's normal operating mode (Rees, 1991). This flexibility makes the SAR widely applicable to missions ranging from one-meter (or better) imaging of "normal" targets (such as buildings or vehicles) to weather pattern, ocean current, and wind velocity sensing and/or tracking (NASA, 1995: sec. 7).

The disadvantages for SAR, as applied to small tactical satellites, include the fact that a SAR requires tremendous amounts of power (typically greater than 500 watts, with still greater peak output requirements for enhanced spotlight and/or boosted signal modes), and the fact that the arrays used for transmission/reception of the microwave signals have to be large (typically 10x2 meters) in order to provide an effective antenna area sufficient to: 1) cover a large swath and 2) exhibit good gain properties. There is a classic design tradeoff between defocusing the antenna for coverage and requiring a sufficient signal to noise ratio for the desired application. Increasing the power output of the SAR also serves to increase the signal to noise ratio of the system, not surprisingly, but at a cost of a higher power requirement placed upon the spacecraft bus. The designer of the antenna must account for many other factors (range of frequencies to be used, polarization, and types of targets to be sensed) as well.

The ultimate application of the mission module drives much of the SAR design, and for a SAR package to be made as small as possible, the designer should focus on a more specialized application. The more specialized the mission the smaller (more optimized) the package may be made (Jack-of-All Trades SAR packages have tended to be quite large -- Canada's RADARSAT antenna with supporting electronics weighs in at a whopping total mass of over 900 kilograms) (NASA, 1995: sec. 7).

A tradeoff in SAR design also exists with the selection of carrier frequency. A SAR operating at higher carrier frequencies is better able to discriminate between phase and frequency shifts associated with the Doppler effect, further enhancing resolution precision; higher and higher microwave frequencies, however, become more and more susceptible to atmospheric attenuation, further increasing noise effects. More signal may be provided by an increase in transmitted power (as stated above), thus combating the effects of atmospheric losses. With these considerations in mind, the tactical SAR mission module should include a high-power (greater than 500 watts) SAR operating at higher-frequencies (Ku, Ka, or X bands, a wavelength range of 1-4 centimeters) and specifically tailored for the tactical military role of all-weather high-spatial-resolution imaging (matching or exceeding the mission and performance requirements for a high-resolution EO mission module). The SAR antenna should incorporate a spotlight mode by utilizing a phased array architecture (saving weight by using fewer mechanical components).

Mass and power estimations for Modsat are based on a lightweight NASA SAR payload under development for flight by the year 2000. This equipment will incorporate an electronically steerable phased array (10x3.5 meters) along with the transmitter/receiver electronics. The drive for this technology development stems from the need for smaller, lighter-weight SAR payloads launchable on small satellites and boosters (*Pegasus, Taurus,* LMLV). The applicability of SAR technology to a multitude of general imaging as well as scientific applications (mentioned earlier) drive this need. The proposed package requires under 500 watts of power and has a mass of under 100 kilograms for both the array and RF driver electronics (NASA, 1995: sec. 7, p. 22). The existence of this technology development makes the small satellite (*Pegasus*-launched) SAR mission module feasible for consideration in this design study. Further mass and volume reductions may be made by specializing the mission module performance to high-resolution imagery, as well as by funding a more aggressive (i.e., military-focused) technology program (the proposed technology program is aggressive in approach, but its focus is on general Earth-science applications).

As with the other payload types, the SAR support requirements are highly dependent upon the intended use of the mission module and the designer's choices. Initial SAR equipment estimation will vary according to the designer's wishes, accounting for the variation in mission module characteristics. Some estimated SAR mission modules (with stowed antenna -- launch configuration) are summarized in Table 13-5.

Table 13-5: Synthetic Aperture RADAR (SAR) Mission Module Estimations

Antenna Dimensions (m x m)	Mass (kg)	Volume (m^3)	Power (w)
8.0 x 1.5	78.4	0.318	800
10.0 x 2.0	86.5	0.564	450

13.4.4 Generally Specified Mission Module Support Requirements

Though the aforementioned mission module types are varied in both type and specification, there are certain requirements on the bus which may be specified, allowing components in many of the spacecraft subsystems to be chosen for all candidate designs. The requirements specifiable through mission module consideration include stabilization control, pointing accuracy, attitude knowledge, thermal isolation, operating power, data handling, data storage, and data down-link.

Data rates of specific mission modules will depend on the number of image pixels processed every second as well as the dynamic range assigned to each pixel. For a basic "pushbroom" linear detector array (electro-optical payload) integrating 1024 (1k) pixels every 130 microseconds (approximately one-meter spatial resolution from a 350 kilometer orbit), the data rate for 16-bit (2-byte) dynamic range is 16.8 megabytes per second (Mbytes/s). Error correction coding effectively doubles the data rate (in the case of the example) to 33.6 Mbytes/s (Brodsky, 1992: p.275). Incorporating 10 spectral channels (across each pixel) into this example calculation (estimating a very high capacity MSI payload), the data rate skyrockets to 336 Mbytes/s. SAR images include not only intensity levels in every image pixel, but also azimuth, elevation, range, frequency, phase, and timing information, amounting to several bytes per reconstructed (synthesized) image pixel. Typical data rates for SAR payloads are above 100 Mbytes/s. The spacecraft bus must be able to accommodate these high data rates. As a minimum, the satellite bus should be able to handle on-board data rates from the payload in excess of 150 Mbytes/sec. Higher on-board data rates may be supported by additional data buffers supplied with the mission module (i.e., higher data rates than those ultimately set by the bus design become "mission specific," and the additional components required for support of these higher rates then become one of the mission module design's requirements).

Data storage capacity requirements will vary according to both the type of mission module, its data collection rate, data compression scheme, and data down-link scheme. A down-link scheme intended to transmit data in near-real-time may require the bus to carry a modest storage device (like the solid state data recorder (SSDR) carried by Clementine 1) of 1 or 2 Gbytes storage capacity. A high data rate mission module or a higher data latency down-link may require substantially more data storage capacity, but the 2 Gbyte SSDR should be considered a minimum. Data compression rates typically average four-to-one, compressing 16-bits into four, but may range up to 12:1 or even 20:1. The tradeoff with data compression schemes is that with increasing compression, the probability of data loss increases. More discussion of data down-linking and data storage may be found in the Command and Data Handling section.

All of the various types of mission modules, all of which are imaging systems, will require a high degree of pointing accuracy, attitude knowledge and stabilization. Depending on the specific mission module, these requirements may differ, but they will all require pointing accuracy on the order of 0.1 degrees and attitude knowledge on the order of 0.05 degrees. As a requirements example, the Clementine 2 spacecraft will be designed to accuracy and knowledge values of 0.05 and 0.03, respectively. Accuracy/knowledge requirements of 0.1/0.05 should be considered maximum allowable margins and should be taken as an attitude system design goal. This range of attitude precision will require a three-axis stabilized spacecraft bus.

Finally, if one bus must be able to support a wide range of mission module types, it must be able to supply sufficient power (both average and peak power requirements). The greatest differences in power requirements may be seen by comparing the passive sensors (EO and MSI) the active sensors (LASER and SAR). The active sensor-equipped mission modules will require substantially more power from the bus than will the passive sensor-equipped ones (several hundred watts for the active sensors compared to about 100 watts maximum for the passive sensors). The highest-demand mission module design will most likely be a SAR-equipped module with a peak power of greater than 500 or 700 watts (for its spotlight mode). The design of the power systems for the spacecraft bus must account for these possibly very high peak power requirements.

In conclusion, all design requirements that the bus must meet will be driven by the most demanding mission module type in all cases. Furthermore, because of the lack of specificity of mission module designs, those driving design requirements must necessarily be interpreted "ranges" of values, as opposed to exact quantities. Table 13-6 summarizes the requirements necessary for support of the various mission module types.

Table 13-6: Estimated Mission Module Support Requirements

Bus Performance Criteria	Mission Module Support Requirement
Pointing Accuracy	0.2-0.1 degrees or better
Attitude Knowledge	0.07-0.05 degrees or better
Data Compression	4:1 minimum
Data Storage Capacity	2Gbytes minimum; modular unit
Data Handling Capacity	150 Mbytes/s or better
Thermal Environment	thermally isolated from mission module
Available Mission Power	peak power from 500-900 watts
Available Mission Launch Mass	120 kg or better
Available Mission Launch Volume	0.6 m^3 or better

13.5 Other considerations

No further changes or refinements were made to the other sections of problem definition. The satellite design team used the data presented in the first iteration for the alterables, constraints, and actors.

14. Value System Design

14.1 The Evolution of the Value System Design

In the first phase the study, the objectives were treated as conceptual attributes by which to subjectively compare the alternatives. As the study progressed to a more detailed level, the team emphasized the need to use measurable objectives as much as possible. In phase two, the initial set of objectives was overhauled in an effort to create a more objective framework for measuring the performance of the alternatives. The team added subobjectives with specific measures of effectiveness that could be obtained through modeling and analysis.

Moreover, the value system was modified to account for feedback from the CDM, advancements in system modeling, and maturity of subsystem engineering. The first phase considered that the main objectives were cost and effectiveness, in order to capture the classic tradeoff between these two attributes. As the study progressed, it was seen that the effectiveness branch of the hierarchy should be decomposed into several main objectives that would capture the various qualities of effectiveness. Availability was retained as a main objective. Responsiveness was re-labeled as tactical responsiveness in order to emphasize the tactical nature of the spacecraft bus. The qualities of flexibility and capability were captured under the new main objective of maximizing mission utility. Finally, the minimization of program risk was added as a main objective.

For the objective of minimizing cost, phase one considered only monetary value. Phase two expanded the notion of cost to consider time and effort. The objective to minimize the time to full rate production was added, while the operations cost subobjectives were evaluated by assessing the effort required to carry out the tasks. Under operations cost, the objective to minimize maintenance cost was added to account for the effort required to keep stored busses and components in mission ready condition.

Under the new main objective to maximize mission utility, several specific performance objectives were added to enhance the evaluation of mission capability.

Under the availability objective, the potential for overlapping attributes was decreased by consolidating the relevant attributes under reliability (natural threats and environmental effects) and survivability (man-made threats).

Under tactical responsiveness, the objective to minimize turn around time to launch was found to cover areas that are beyond the scope of this study, such as launch vehicle operations, operational procedures, and issues regarding the structure and manpower of the necessary operational Therefore, this objective was changed to the objective of minimizing preparation time to launch, in order to capture those elements of the alternative concepts that are relevant to the study and affect this important attribute.

The responsiveness sub-objective of minimizing time for data was changed to the objective of minimizing of data latency. The original definition included elements that were beyond the scope of this study, such as ground communication and ground processing. The new definition concentrates only on the relevant aspects of the spacecraft bus, such as data processing and data down-linking architectures.

The objective to maximize the capability for tactical maneuvers was added under the tactical responsiveness objective, in order to emphasize the desire of the CDM for a tactical platform.

Under the new objective of minimizing program risk, sub-objectives were added to cover the areas of cost, schedule, and performance risk.

14.2 Modsat Objectives

The set of objectives used for this study is intended to fully capture the values of the CDM and provide a consistent framework for the evaluation of alternatives. Thus, all of the objectives discussed below were used to guide the development of subsystem and system-level solutions. Throughout the design of the overall system, many tradeoffs were performed at the system and subsystem levels, in an effort to form a reasonably sized solution space within which alternative system solutions could be evaluated. As a result, the remaining alternative solutions satisfy several of the objectives equally well. Although these objectives do not contribute to the determination of the best solution, they were instrumental in the development of each candidate.

The main objective of the Modsat study was to produce the best standardized bus for small, tactical, low-earth orbit satellites. The top-level objectives are shown in Figure 14-1. The following sections explain each objective.

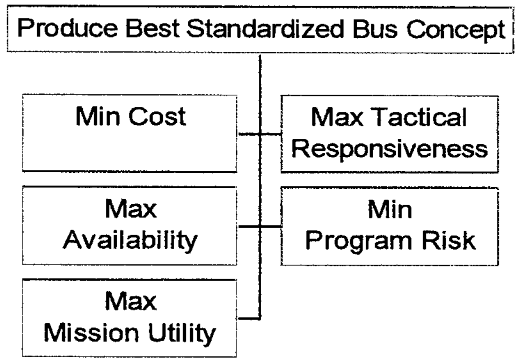


Figure 14-1: Top-level Objectives

14.3 Measures Of Effectiveness

Each bottom-level sub-objective has a unique measure of effectiveness. The ideal MOE is a natural scale which can be directly measured or computed, such as speed in meters per second. However, the true MOE is often difficult and impractical to obtain. This is especially true for studies which occur early in the life-cycle of a system, where modeling and testing is limited. In this case, two options are available (Clemen, 1996:79). The first is to use a proxy measurement. The proxy should be closely related to the objective under consideration. The second option is to "construct an attribute scale for measuring achievement of the objective" (Clemen, 1996:79). This requires the definition of levels of performance for the objective, with levels ranging from best to worst. According to Clemen,

> The key to constructing a good scale is to identify meaningful levels, including best, worst, and intermediate, and then describe those levels in a way that fully reflects the objective under consideration. The descriptions of the levels must be elaborate enough to facilitate the measurement of the consequences (Clemen, 1996:80).

Several of the MOEs for this study are actually combinations of proxy measurements and attribute scales. For instance, all sub-objectives of the objective to minimize pre-launch operations cost are measured by an attribute scale that determines the difficulty of performing the task, as opposed to the actual dollar cost of performing the task. All attribute scales used for this study have six levels, from zero (worst) to five (best). It was felt that six levels provided enough detail to make intelligent judgments. Also, the scale from zero to five translates well into the common utility scale of zero to one (discussed in section 14.5).

The MOEs for each objective are described in section 14.4. Included for several of the objectives are the contributing factors that aided in the construction of attribute scales. For some of the attribute scales, it was not necessary to fully define all six levels, since the definition of the best and worst levels made it clear how the intermediate levels would be defined.

Although most of the MOEs are constructed attribute scales, future design efforts for this program must convert these to natural scales wherever possible.

14.4 Objective Hierarchy

14.4.1 Minimize Cost

The sub-objective hierarchy under "Minimize Cost" is shown in Figure 14-2. For this study, cost is made up of two main elements. The first element is monetary cost. The performance of each monetary cost objective may not be measured in actual dollars; for some objectives a proxy utility scale will describe the cost. For others, cost estimating relationships (CERs) may be used where such relationships are available. CERs yield a cost in dollars, but all such costs must be stated for the same year (i.e., FY96 dollars).

The second main element of cost is time to full rate production.

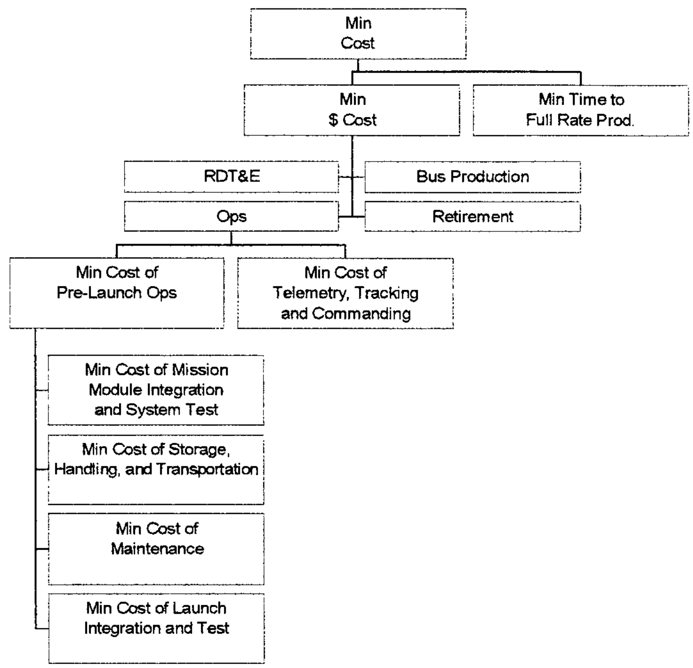


Figure 14-2: Cost Objectives

14.4.1.1 Minimize Monetary Cost

Monetary cost has several elements, discussed below. Some key points that were kept in mind when evaluating costs for this study were:

- Avoid overlapping and double-counting. Without careful accounting, the various elements of development, production, and operations cost can be easily counted more than once under different calculations. This would inflate the overall estimation of cost, and would tend to make the cost objective appear more critical than it would be otherwise.
- Use the same fiscal year base for dollar values.

- The use of any CER must be validated as to its applicability to the problem. The use of irrelevant CERs can lead to serious errors in cost estimation For example, CERs for the radio equipment of different aircraft types may be similar, but the engine CERs for a bomber and a fighter could be quite different. The two types of engines are designed to satisfy very different demands. Thus, the use of bomber engine CERs for fighters could cause inaccurate estimations.
- This preliminary design study is not an appropriate forum for a highly detailed cost analysis. Any generated cost values were intended for comparing the relative program element costs of competing alternative solutions, and should not be interpreted as actual predicted costs for the spacecraft bus.
- Cost values for recurring and non-recurring costs are not additive, and must not be added together in attempt to find an overall cost. Alternative solutions will be evaluated separately for each attribute.

There are four main sub-objectives under "Minimize Monetary Cost":

14.4.1.1.1 Minimize Research, Development, Test and Engineering (RDT&E) Cost

This category includes all non-recurring costs from the beginning of the project to the production effort.

MOE: Cost estimating relationship

14.4.1.1.2 Minimize Bus Production Cost

This objective covers the cost of manufacturing the final product.

MOE: Cost estimating relationship

14.4.1.1.3 Minimize Retirement Cost

A complete spacecraft design effort should consider disposal of the spacecraft at the end of its useful life. The spacecraft could be retired via reentry into the atmosphere (by natural orbital decay or by a ΔV maneuver), or by placement into a "retirement" orbit. The method of retirement will dictate the cost. For instance, a ΔV burn-in requires the reservation of propellant for retirement purposes. On the other hand, it can be done quickly. A natural decay retirement requires no propellant, but will tap space operations resources, as personnel and equipment must be used to keep track of the useless spacecraft.

MOE: Attribute scale

Contributing factors:

- Amount of excess fuel for retirement purposes
- Time required to track the dead satellite
- Hazards to environment/people

5 Satellite can be retired quickly after its useful life is gone; retirement procedure is simple; no significant hazards to environment/people.

.
.
.

0 Dead satellite will remain in orbit for years before retirement; complex procedures; significant hazards to environment/people.

14.4.1.1.4 Minimize Operations Cost

This cost can be divided into pre- and post-launch costs.

14.4.1.1.4.1 Minimization of Pre-Launch Operations Cost

This is the sum of all ground expenditures necessary to prepare the satellite for launch. Since this study occurred as a first step in the potential life-cycle of the tactical bus program, the performance of the pre-launch operations cost subobjectives cannot be realistically measured in dollars. Rather, they can each be evaluated by proxy attributes. The sub-objectives for this cost are:

14.4.1.1.4.1.1 Minimize Cost of Mission Module Integration and System Test

This cost covers all efforts to mate the mission module to the bus, and to test the integrated satellite. There are various manpower, equipment, software, and overhead costs associated with connecting the power, signal, thermal, and structural interfaces. The testing effort includes all actions to ensure the integrated satellite is ready to be shipped for launch.

Proxy: Maximize Ease of Mission Module Integration and System Test

MOE: Attribute scale

Contributing factors:

- Complexity of bus/mission module interface
- Complexity of total spacecraft
- Number of components and parts
- Accessibility of components and connections
- Number of connections and cables
- Number of possible configurations
- Complexity of signal flows (power, data)

5 Easy to integrate and test; simple bus/mission module interface; few components and parts; few cables and connections; few subsystem interfaces; total accessibility of all components for diagnostic testing; one standard bus configuration.

.

.

0 Extremely difficult to integrate and test; highly complex bus/mission module interface; many components and parts; abundant and complicated system of cables and connections; components are highly inaccessible; many possible system configurations; complex system of signal flows.

14.4.1.1.4.1.2 Minimize Cost of Storage, Handling, and Transportation

Includes facilities, equipment, special procedures, manpower and overhead.

Proxy: Maximize Ease of Storage, Handling, and Transportation

MOE: Attribute scale

Contributing factors:

- Number of sensitive parts needing special care while storing, handling, and transporting
- Amount and complexity of special equipment required to store, handle, and transport
- Special requirements for storage, handling and transportation (i.e., clean room requirements)
- Number and severity of required safety precautions (i.e., hydrazine propellant is very hazardous)
- Amount of storage time required for parts and assembled busses.
- Classification of program

5 Standard clean room practices for storage and assembly; no extra sensitive parts; minimum basic safety precautions; no unusual equipment required to handle parts; no special classification difficulties; standard inventory-supply system to track and store parts and assembled busses; minimum storage time required; no unusual transportation requirements.

.

.

.

0 Very strict storage requirements (vacuum chamber, nuclear durable, etc.); many highly sensitive components; many complicated safety precautions; highly unique procedures and equipment required; highly unique inventory-supply system required; special classification difficulties.

14.4.1.1.4.1.3 Minimize Cost of Maintenance

There are various costs associated with maintaining the spacecraft hardware and software, including manpower, spares, and supplies. This covers both periodic maintenance and repairs. Since the operations concept includes the storage of several busses, periodic maintenance checks will be necessary to support the equipment. This is especially true for those components that may have a relatively short shelf-life.

Proxy MOE: Maximize Ease of Maintenance

MOE: Attribute scale

Contributing factors:

- Amount of maintenance required
- Accessibility of components
- Cost and Availability of parts
- Subcontractor warranty policies
- Shelf of life parts
- Expected storage time of unassembled components
- Expected storage time of assembled components
- Complexity of design

5 Minimum inventory; all parts ordered just prior to assembly; assembled bus quickly integrated with mission module and shipped for launch; highly robust parts and design, requiring very little routine maintenance; simple bus configuration, with easy accessibility to components; components readily available and not overly expensive; components under warranty; infinite shelf-life of parts.

4 Small inventory; some storage time required for parts and assembled busses; robust parts and design; some routine maintenance; simple-to-moderately complex bus design; easily accessibility to components; components readily available and under warranty; long shelf-life.

3 Small-to-moderate inventory; moderate storage times required; some routine maintenance; moderately complex design; some accessibility to components; components available with short lead times; moderate shelf-life.

2 Moderate inventory; moderate-to-long storage times; much routine maintenance; complex design; difficult to access components; moderate lead times; marginal shelf-life.

1 Large inventory; long storage times; much routine maintenance; complex design; very difficult to access components; long lead times; short shelf-life.

0 Large inventory, with many components held for a long time; long-lead time required for delivery of parts; expensive parts; assembled bus held in storage for many months; parts break before needed, and not covered under warranty; complex bus design requiring extensive routine maintenance; components highly inaccessible.

14.4.1.1.4.1.4 Minimize Cost of Launch Integration and Test

This covers all efforts to mate the satellite to the launch vehicle, and to test the loaded rocket.

Proxy: Maximize Ease of Launch Integration and Test

MOE: Attribute scale

Contributing factors:

- Complexity of LV-to-spacecraft interface
- Sensitive parts requiring special care and precaution, such as spacecraft propellants, solar arrays, antennas, sensors, etc.
- Special integration considerations
- Number of connections for power and data flow
- Difficulty of testing the integrated spacecraft
- Rigidity of structure

5 Simple interface; few connections and attachment points; rigid, solid structure; few sensitive parts and special precautions required; few/small deployable structures (solar arrays, antennas, etc.); maximum accommodation of testing; few unique integration issues.

.
.
.

0 Highly complex interface; many connections and attachment points; awkward structure, susceptible to undesirable moments, forces and frequencies; many sensitive parts and systems, requiring many special

precautions and procedures; awkward and inhibiting deployable structures; testing the integrated system is difficult; several unique integration issues.

14.4.1.1.4.2 Minimize Cost of Telemetry, Tracking, and Commanding (TT&C)

This cost covers all operational expenditures for tracking, commanding, and maintaining the orbiting satellite. Elements of TT&C include ground segment hardware and software, satellite operations personnel, and communications and data flow operations. Although the evaluation of the most of these elements is beyond the scope of this study, alternative concepts will be judged as to how well they facilitate the TT&C function.

MOE: Attribute scale

Contributing factors:

- On-board processing
- Dedicated antennas, number of antennas
- AFSCN compatible
- Data down-link
- Data delivery

5 All data processing can be done on-board; fully autonomous; AFSCN compatible; high data rate transmission on separate band; user-friendly, simple but powerful system for commanding and analyzing telemetry.

4 Most of the data processing can be done on-board; highly autonomous; AFSCN compatible; high data rate transmission on separate band; user-friendly, simple system for commanding and telemetry, with standard analysis features.

3 Some on-board data processing; semi-autonomous; AFSCN compatible; high data rate transmission on separate band; user-friendly, simple system for commanding and telemetry, with standard analysis features.

2 Some on-board data processing; no autonomy; AFSCN compatible; standard command and telemetry system.

1 No on-board data processing; no autonomy; AFSCN compatible; much ground support required; standard command and telemetry system.

0 No on-board processing; no autonomy; extensive ground support required, with many man/computer hours; only compatible with a single, dedicated ground station (no AFSCN compatibility).

14.4.1.2 Minimize Time to Full Rate Production

Assuming that this program will be approved for further development at each program milestone, the time required to achieve full rate production of busses should be minimized.

MOE: attribute scale

Contributing factors:

- Complexity
- Level of technology
- Availability of technology
- Challenges to manufacturing

5 All technology currently available and on the shelf; no special development effort required; no long-lead times for parts and processes; no significant advances in manufacturing required.

4 All technology currently available; one or two significant challenges in development; no major manufacturing advances required.

3 Most technology currently available; a few significant challenges in development; at least one major manufacturing advance required.

2 Most technology currently available; many significant challenges in development; one or two major manufacturing advances required.

1 Some technology currently available; many significant development challenges; a few major advances in manufacturing required.

0 Much technology under development or unavailable; most parts and processes require special development effort; several advances in manufacturing technology required.

14.4.2 Maximize Tactical Responsiveness

Modsat is intended for tactical applications, as has been stressed by the CDM. Thus, responsiveness is a primary objective. Modsat satellites must be able to respond quickly to rapidly generated needs and mission requirements. One major driver of responsiveness is the availability of a launch vehicle; however, it was assumed for this study that launch vehicles are continuously available.

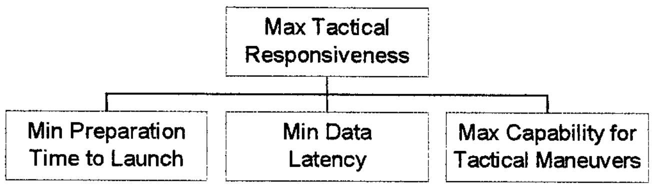


Figure 14-3: Tactical Responsiveness Objectives

14.4.2.1 Minimize Preparation Time to Launch

This time interval begins with the demand for a particular Modsat mission, and ends with the delivery of the satellite for launch vehicle integration. The phrase "launch-on-need" has been used throughout the study. It implies the ability to launch within a few days, as opposed to several months.

MOE: Attribute scale

Contributing factors:

- Number and level of activities required to integrate mission module to bus and prepare satellite for launch
- Number of man-hours required to integrate mission module to bus and prepare satellite for launch
- Number of equipment/computer hours required
- Special systems and processes required
- Safety hazards
- Complexity of bus-to-mission module interface

- Complexity of launch vehicle-to-payload interface
- Number and complexity of integrated system tests

Assumptions:

- Launch vehicles and launch sites are available when the need arises
- Each alternative uses the same manpower and facility scheduling scheme

5 Relatively few activities required to integrate the mission module to the bus, and to prepare the satellite for launch; process is not man-hour or equipment-hour intensive; few special systems and processes required; relatively few significant safety hazards exist; simple interfaces; only a few, simple tests required on the integrated systems.

.

.

.

0 Many difficult activities required to integrate the mission module to the bus, and to prepare the satellite for launch; process is highly man-hour and equipment-hour intensive; many special systems and processes required; many significant safety hazards exist; highly complex interfaces; many challenging tests required on the integrated systems.

14.4.2.2 Minimize Data Latency

Data latency refers to the time between mission data collection and reception by the user in raw form. This objective should capture the differences in data processing and data down-linking architectures.

MOE: Attribute scale

Contributing factors:

- Down-link rate.
- On-board data processing capability.
- Data delivery architecture.

5 Very fast turnaround-time from data collection to data reception in useable form; highly flexible and capable data delivery architecture, compatible with AFSCN, dedicated and portable stations; very high data down-link rate.

4 Fast turnaround-time from data collection to useable reception; highly capable and somewhat flexible data delivery architecture; very high data down-link rate.

3 Moderate turnaround-time from data collection to useable reception; highly capable but inflexible data delivery architecture; high data down-link rate.

2 Moderate turnaround-time from data collection to useable reception; moderately capable, inflexible data delivery architecture; moderate down-link rate.

1 Slow turnaround-time from data collection to useable reception; low-capability, inflexible data delivery architecture; moderate to low down-link rate.

0 Very slow turnaround-time from data collection to useable reception; very low-capability, highly inflexible data delivery architecture; low to very low down-link rate.

14.4.2.3 Maximize Capability For Tactical Maneuvers

The satellite may be called on to perform plane changes or slewing maneuvers in response to tactical mission needs. This sub-objective covers both slewing capability and ΔV (change in velocity) capability, which depend on how much extra fuel is loaded beyond that required for normal usage. ΔV is a more useful measure than the mass of extra fuel, since ΔV performance is proportional to specific impulse, which can be different for different fuels and propulsion systems of the same weight.

MOE: Attribute scale

Contributing factors:

- Slew rate
- Amount of delta-V (fuel) available for plane changes

Note: The output torque capability of the reaction wheels is a proxy measure for the achievable slew rate. The available delta-v for plane changes is found by subtracting from the total delta-v supply the amount needed for altitude maintenance and momentum dumping.

5 Very fast slew rate; two-degree plane change allowed.

4 Fast slew rate; 1.5-degree plane change allowed.

3 Moderate slew rate; one-degree plane change allowed.

2 Moderate slew rate; 0.5-degree plane change allowed.

1 Slow slew rate; 0.5-degree plane change allowed.

0 Very slow slew rate; no capability for plane changes.

14.4.3 Maximize Availability

A sound system design for Modsat must attempt to maximize on-orbit availability. The bus must endure both natural and man-made hazards. Natural hazards are considered under reliability, while man-made hazards are considered under survivability.

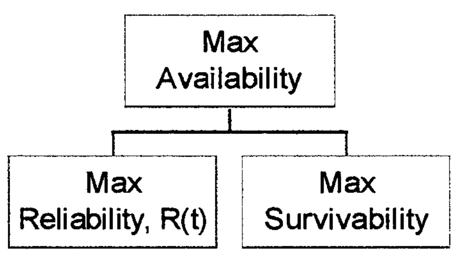


Figure 14-4: Availability Objectives

14.4.3.1 Maximize Reliability

For the purposes of this study, reliability refers to the probability that, given a non-hostile environment, the bus will be able to perform its primary mission of supporting the mission module at a given point in its lifetime. It is partly a measure of the hardness of the bus to the natural space environment over the satellite's lifetime, within a specified confidence level. Since the maximum lifetime requested by the CDM was one year, this objective was as evaluated as the reliability at one year. But high reliability beyond the one year point has value, in that more use could be made of a given Modsat satellite. If it could be shown that Modsat could achieve high reliability at two or even three years, with little impact on cost, this would greatly interest the CDM.

MOE: Overall system reliability at one year, R(one year).

14.4.3.2 Maximize Survivability

Survivability refers to the ability of the system to perform its intended mission after exposure to stressing environments created by enemy or hostile agent.

MOE: Attribute scale

Contributing factors:

- "Safe" modes of operation
- Ability for evasive maneuvers
- Shielding from radiation and EMP
- Protection of circuitry and computer memory

5 Auto safe mode; full array of evasive maneuvers available; structure and components have maximum shielding against radiation and EMP; extensive protection of circuitry and computer memory.

4 Remote safe mode; limited evasive maneuvers; structure and components have moderate level of shielding; extensive protection of circuitry and computer memory.

3 Remote safe mode; limited evasive maneuvers; structure and components have moderate level of shielding; limited protection of circuitry and computer memory.

2 Limited remote safing actions; no evasive maneuvers; limited shielding of structure and components; limited protection of circuitry and computer memory.

1 No safe mode; no evasive maneuvers; no shielding of structure and components; limited protection of circuitry and computer memory.

0 No safe mode; no evasive maneuvers; no shielding of structure and components; no protection of circuitry and computer memory.

14.4.4 Minimize Program Risk

Program risk refers to the potential for elements of the program to fail to come together as planned. Risk can be assessed in the areas of cost, schedule, and performance.

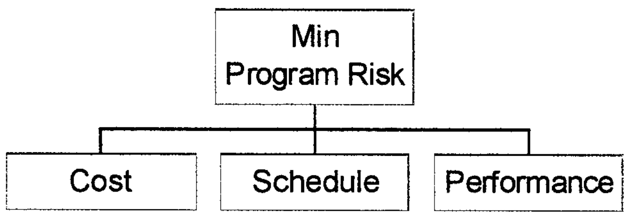


Figure 14-5: Program Risk Objectives

14.4.4.1 Minimize Cost Risk

Each system development program has cost risk, in that the predicted life cycle costs or individual elements of the cost may be much higher than planned. In a sense, cost risk is a measure of the lack of confidence in the cost estimates. An accurate and dependable cost model helps to lower cost risk.

Note the difference between cost risk and cost. An alternative solution that has a high cost but for which there is a great deal of confidence in the cost estimate would have low cost risk. Likewise, an alternative that has a low cost but for which there is low confidence in the cost estimate would have high cost risk.

MOE: Attribute scale

Contributing factors:

- Uncertainty in cost estimates
- Unique systems, parts, and procedures

5 Exact knowledge of cost of all facets of bus recurring and non-recurring costs.

4 Knowledge of costs for many aspects of the program; all systems, parts, and processes have much historical cost data; CERs are well-suited to estimate costs for all areas where exact knowledge doesn't exist.

3 All systems, parts, and processes have much historical data; CERs are well-suited to estimate costs for most areas; some proxy measures instead of CERs.

2 Most systems, parts, and processes have historical data; most CERs are well-suited to estimate costs; some uncertainty in estimates; some proxy measures used instead of CERs.

1 Few systems, parts, and processes have historical data; most CERs are not well-suited to estimate costs; many proxy measures used instead of CERs; much uncertainty in estimates.

0 Total uncertainty in cost estimates; CERs are not-well suited; most systems, parts and processes do not have historical data; almost all objectives must be measured with proxy measures.

14.4.4.2 Minimize Schedule Risk

Schedule risk refers to the potential for unforeseen slips in the development schedule. Schedule risk can be mitigated with sensible planning and accurate forecasting.

MOE: Attribute scale

Contributing factors:

- Unique, exotic equipment and processes required in manufacturing, test, integration, and support
- Amount and level of unproved technology used on the bus
- Difficulty of program planning and prediction

- Potential for tricky integration issues, both for the mission module and the launch vehicle

5 Few unique systems and processes required in manufacturing, test, integration, and support; all technology is flight-qualified and well-proven; no major foreseen difficulties in program planning and prediction; no major foreseen tricky integration issues.

.
.
.

0 Most systems and processes for manufacturing, test, integration, and support are highly unique; much unproved, cutting-edge technology; several major foreseen difficulties in program planning and prediction; several major foreseen tricky integration issues.

14.4.4.3 Minimize Performance Risk

There is a chance that some technological aspect of the system, be it hardware or software, will not work as well as planned. This usually depends on the maturity of the technology in question.

MOE: Attribute scale

Contributing factors:

- Amount and level of unproved technology used on the bus
- Amount and level of testing

5 Almost no cutting-edge technology; all hardware and software is flight-qualified and well-proven; extensive test program.

.
.
.

0 Much cutting-edge technology; very little flight-qualified hardware and software; very little testing achieved or even possible.

14.4.5 Maximize Mission Utility

Mission utility refers to the ability of Modsat to accommodate a range of different mission modules. In other words, it is a way of quantifying how well the bus performs its role of being generic and standard. The larger the range of possible missions, the higher the mission utility. This objective was difficult to construct, since it is hard to envision how such utility can be measured. It was decided that mission utility is supported by various aspects of spacecraft bus performance such as pointing accuracy and available power. In other words, if more performance capability is built into the bus, more mission types can be accommodated. Thus, several performance sub-objectives were adopted, in addition to the obvious desire to maximize the available weight and volume for the mission module.

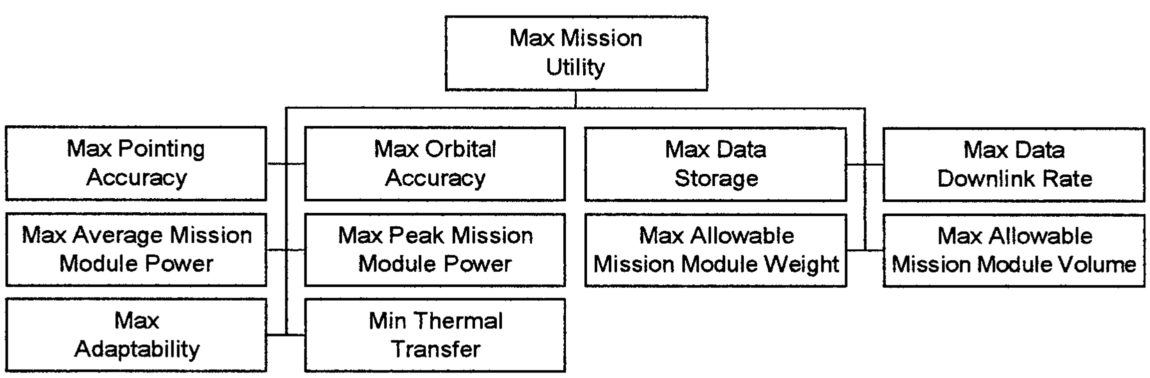


Figure 14-6: Mission Utility Objectives

14.4.5.1 Maximize Pointing Accuracy

Many space mission applications require a great deal of spacecraft pointing accuracy, for precise pointing and orientation of sensing instruments.

MOE: Degrees of pointing accuracy (smaller values are desired)

14.4.5.2 Maximize Orbital Accuracy

The satellite must be able to maintain its intended orbit, in order to be available for its intended applications.

MOE: Attribute scale

Contributing factors:

- Whether or not there is a propulsion system on-board
- Amount of fuel, in terms of delta-v, reserved for maintaining orbital altitude

5 Enough propellant to maintain the full orbital altitude for the design life of the satellite.

.

.

.

0 No propellant reserved for orbital maintenance.

14.4.5.3 Maximize Data Storage

Various missions may require the spacecraft processor to temporarily store mission data for later transmission.

MOE: Attribute scale

Contributing factors:

- Size of data recorder (gigabits of storage)
- Capability of data recorder (compression, etc.)
- Rate of data collection
- Frequency of data transmission

5 Very large, highly capable device; maximum compression; handles toughest scenario of high rate of collection but low frequency of transmission; can store high resolution data of several orbits.

4 Large, very capable device; high compression; handles high rate of collection and moderate frequency of transmission; can store high resolution data of more than one orbit.

.

.

.

1 Small recorder with minimum capability; no compression; handles only low rate of collection and data must be transmitted quickly; stores low resolution data of a short pass within an orbit.

0 No flight data recorder on-board; zero capability for data storage.

14.4.5.4 Maximize Data Down-link Rate

Although Modsat must be compatible with the Air Force Satellite Control Network (AFSCN), some missions may require a higher data down-link rate than can be accommodated with AFSCN compatible communications equipment. Systems solutions that provide for higher rates would be more attractive, all other factors being equal.

MOE: Mbytes/sec

14.4.5.5 Maximize Average Mission Module Power

This objective strives to maximize the amount of power, on average, that Modsat makes available to the mission module. A survey of existing LEO military satellites reveals that their power requirements cover a wide range.

MOE: Watts of available average power

Note: Available average power is the total average power minus the total power required by the bus.

14.4.5.6 Maximize Peak Mission Module Power

Some mission applications will require large amounts of power during peak operations. Modsat must be able to handle peak demands which can greatly exceed the average demand.

MOE: Watts of available peak power

Note: Available peak power is the total peak power minus the total power required by the bus.

14.4.5.7 Maximize Allowable Mission Module Weight

One of the main design goals of this study is a lightweight bus which allows for the heaviest possible mission module. The amount of allowable mission module weight is determined by subtracting the bus weight from the total allowable weight, which depends on the launch vehicle and the selected orbit.

MOE: Kilograms of available weight

Note: Allowable mission module weight is defined as the weight that can be launched to the specified orbit minus the total wet weight of the bus.

14.4.5.8 Maximize Allowable Mission Module Volume

This sub-objective is met by minimizing the volume of the bus. The amount of allowable mission module volume is determined by subtracting the bus volume from the total allowable volume, which depends on the volume and dimensions of the launch vehicle payload fairing.

MOE: cm^3 of available volume

Note: Allowable mission module volume is defined as the usable volume of the launch vehicle minus the total volume of the bus.

14.4.5.9 Maximize Adaptability

This sub-objective should capture the CDM's desire for flexibility of design and adaptability to changing mission requirements. The ability to add needed capability or remove excess capability is valued, while at the same time consideration must be given to minimizing the amount of engineering which must be applied to the integration effort. Alternative solutions will be judged on how well they meet this goal.

MOE: Attribute scale

Contributing factors:

- Degree of modularity and flexibility in the bus
- Ability to insert more capability or remove excess capability
- Breadth of mission profile capability
- Ease of updating subsystems and components with technological upgrades, without a major system redesign

5 Supports a wide breadth of mission profiles; highly flexible design with near total modularity; nearly all subsystems can be easily replaced with new versions; additional capability can easily be inserted; excess capability can easily be removed.

4 Supports several different mission profiles; flexible design with much modularity; many subsystems can easily be replaced with new versions; some additional capability can be inserted; some excess capability can be removed, but not all.

3 Supports several different mission modules with a few different mission profiles; moderate amount of modularity and flexibility; a few subsystems/components can be replaced without major redesign; some excess capability can be removed; limited ability to add additional capability.

2 Supports a few different mission modules with one or two different mission profiles; limited modularity and flexibility; only one or two subsystems/ components can be replaced without major redesign; limited ability to remove/add capability.

1 Supports only similar mission modules with the same mission profile; limited modularity or flexibility; fixed design, with almost all component

upgrades requiring a major system redesign; most components are fixed in structure, with limited capability for removal or replacement; does not support additional capabilities.

0 Supports only a single baseline mission module; virtually no modularity or flexibility; fixed design, with almost all component upgrades requiring a major system redesign; all components are fixed in the structure, with no capability for removal or replacement; does not support additional capabilities.

14.4.5.10 Minimize Thermal Transfer

It is necessary to limit the amount of heat energy that passes between subsystems and components, as well as from the bus to the mission module. Many components on a spacecraft can be very sensitive to heat. Consideration must be given to the highly dynamic nature of a spacecraft thermal environment.

MOE: Attribute scale

Contributing factors:

- Amount of heat energy crossing the interface between bus and mission module
- Quality of thermal management within the bus

5 Virtually no heat energy crosses the interface; excellent thermal management on the bus.

.
.
.

0 Much heat energy crosses the interface; poor thermal management on the bus.

14.5 Utility Functions

In order to provide overall utility scores for each competing system solution, all MOEs must be converted to a common utility scale. A utility scale from zero to one was chosen for convenience, but the endpoints of the scale could be any numbers. The translation from MOE to utility for a given objective is referred to as the utility function of that objective. A utility function is essentially a model that represents the preferences of the CDM for an objective (Clemen, 1996:473). According to Clemen, "assessing a utility function is a matter of subjective judgment" (Clemen, 1996:473). Feedback from the CDM enables the analyst to determine how much utility to assign a given level of performance.

14.5.1 Theory

Figure 14-7 shows the three basic shapes of utility functions (Clemen, 1996:466). Often, a higher level of performance, or payoff, is associated with greater risk. In other words, the price of trying to achieve higher performance is a greater probability of a failure or loss of some kind. For example, a gambler may have to accept a greater risk of losing money if he wants a chance to win a big payoff.

An individual with risk-averse behavior will accept a lower payoff in return for less risk of loss. For this individual, there is a diminishing return on increases in the level of performance. Thus, a risk-averse utility curve is concave (Clemen, 1996:465). The CDM may be risk-averse (diminishing return on performance) with regard to several of the objectives. For other objectives, the CDM may feel that maximizing performance is the same as maximizing utility, so that a linear translation of payoff to utility is appropriate. This is known as a risk-neutral utility function, and the corresponding utility curve is a straight line (Clemen, 1996:466). It is possible that the CDM may be risk-seeking with regard to an objective. A risk-seeker will look for a high payoff, regardless of the risk. The corresponding utility curve is convex (Clemen, 1996:465).

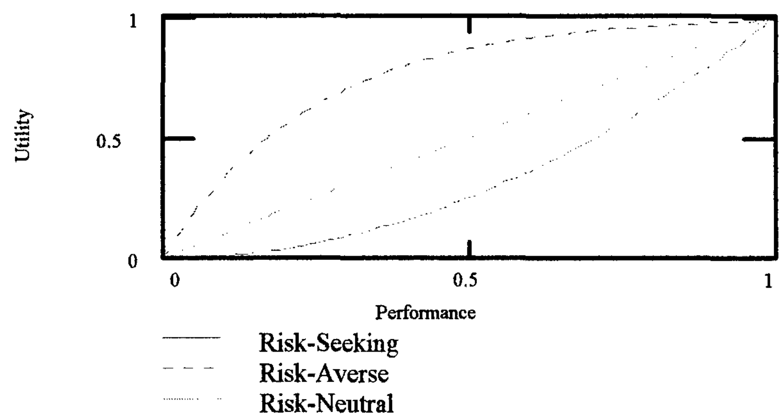


Figure 14-7: Basic Utility Functions

14.5.2 Application

Ideally, each objective would have a unique utility function that reflects the risk attitude of the CDM. However, in the absence of participation from the CDM, the team took a generalized approach. It was felt that the main objective areas of cost and risk should have risk-averse utility functions, with the exception of RDT&E and production costs (discussed below), while the main areas of tactical responsiveness, availability, and utility should have risk-neutral utility functions. This was a subjective judgment, based on the combined experience and knowledge of the team members. The validity of the results of this study could be improved with feedback from the CDM with regard to utility functions.

All of the objectives with risk-averse utility functions have attribute scales from zero to five. To be conservative, the team chose a slightly concave curve, with a translation as shown in Table 14-1.

Table 14-1: Risk-averse Utility Conversion

Performance	0	1	2	3	4	5
Utility	0.00	0.30	0.55	0.75	0.90	1.00

The corresponding utility curve is shown in Figure 14-8. During the analysis phase of this study, non-integer values of performance were translated using a curve-fitting function.

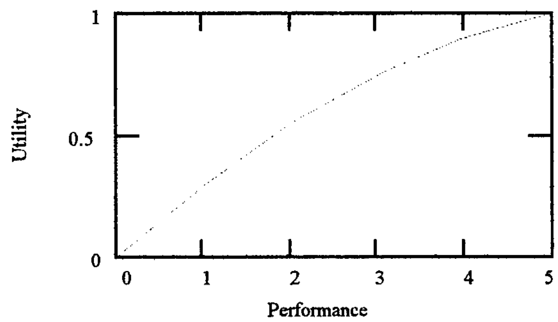


Figure 14-8: Risk-averse Utility Curve

All risk-neutral objectives have linear translations from performance levels to utilities. For those objectives with attribute scales as MOEs, the utility is found by dividing the performance level by five. All objectives in this study that have natural MOEs are risk-neutral. For the sake of simplicity, RDT&E and production costs were included in this group.

The assignment of utilities for natural MOEs is somewhat more difficult than for attribute scales. The analyst must decide what level of performance receives a perfect utility score of one, and what level receives the worst score of zero. Two approaches are possible. On one hand, the analyst can determine the absolute best and worst values that could be achieved within the context of the problem. This approach relies on the knowledge of subject matter experts who are familiar with the problem, and requires much research on the part of the analyst. On the other hand, the analyst can wait to assign utilities until the performance values of all the alternative solutions have been recorded. In this scheme, the best value receives a utility of one, the worst value receives a utility of zero, and all values in-between are proportionally scored between zero and one. Given the time constraint and lack of space design experience on the team, the latter approach was chosen.

14.6 Priority Weighting of The Objectives

As in phase one of the study, objectives on the same level in the hierarchy were assigned priority weights, as shown in Figure 14-9. These weights were determined in the same manner as in phase one, based on the results of a preference chart survey. This survey was completed by the CDM, members of the team, and other subject matter experts, with feedback from the CDM being more heavily weighted. The actual survey is attached as Appendix A.

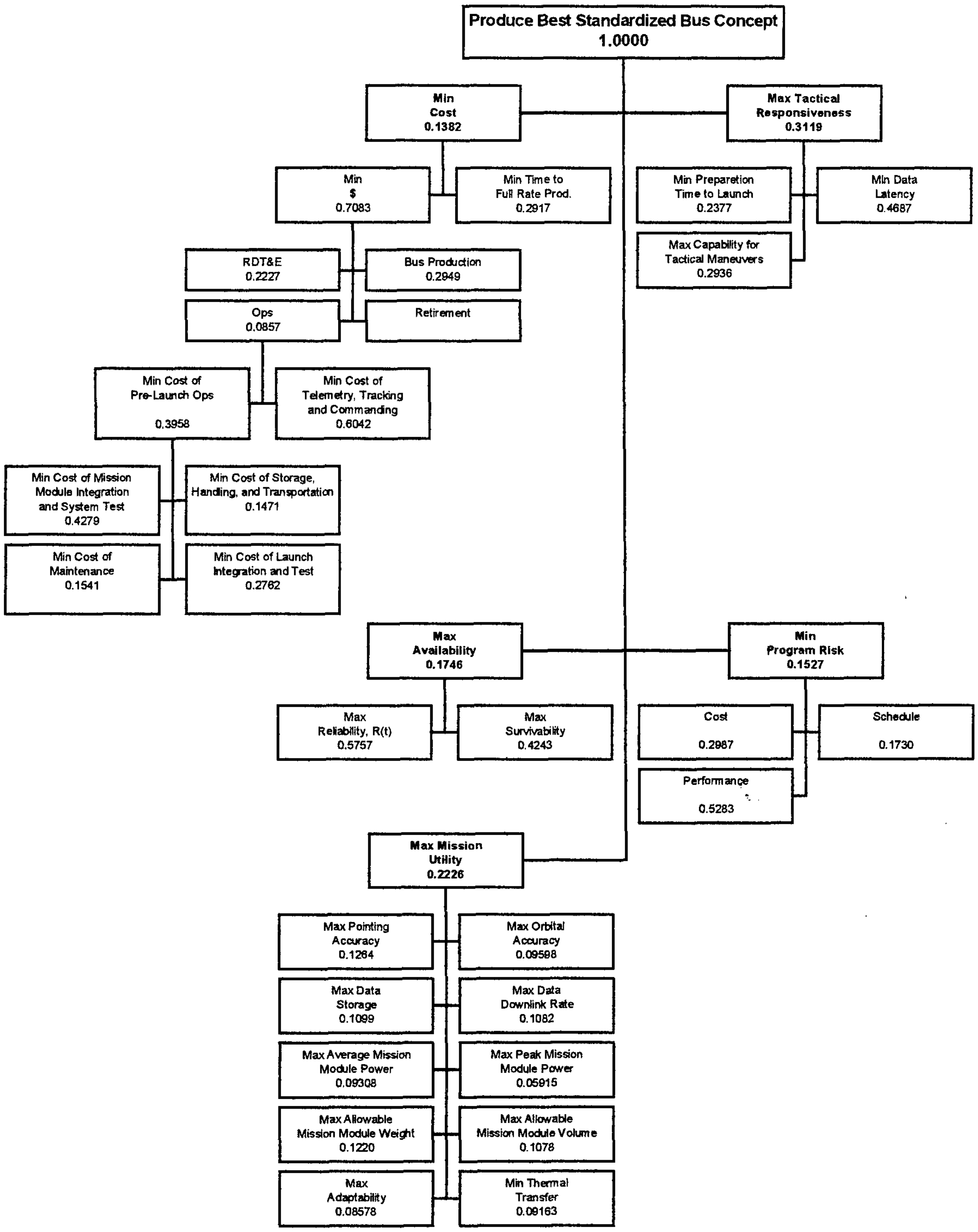


Figure 14-9: Phase Two Objective Hierarchy

From Figure 14-9, it is clear which objectives are more important than others in the current value system. It should be noted that this set of weights is a reflection of personal opinions, solicited at a certain time and under a given set of technological, political, and economic conditions. Major changes in any of these areas could cause the relative priorities to change. This potential for change does not present a significant problem, since the overall scoring function (see section 14.7) can easily be re-calculated with new weights. In fact, a sensitivity analysis was performed on the alternative solutions by varying the weights of each of the top-level objectives. See "Decision Making" for the results of this analysis.

A comparison of the priorities of the top-level objectives is shown in Figure 14-10. Tactical responsiveness is the highest-rated objective, while mission utility has the second highest priority. It is interesting to note that the cost objective received the lowest rating, as opposed to its prominence in most system studies.

Figure 14-9 shows that the minimization of dollar cost is far more important than the minimization of the time to full rate production. Within the dollar cost objective, retirement cost has the highest priority, while operations cost has the lowest priority.

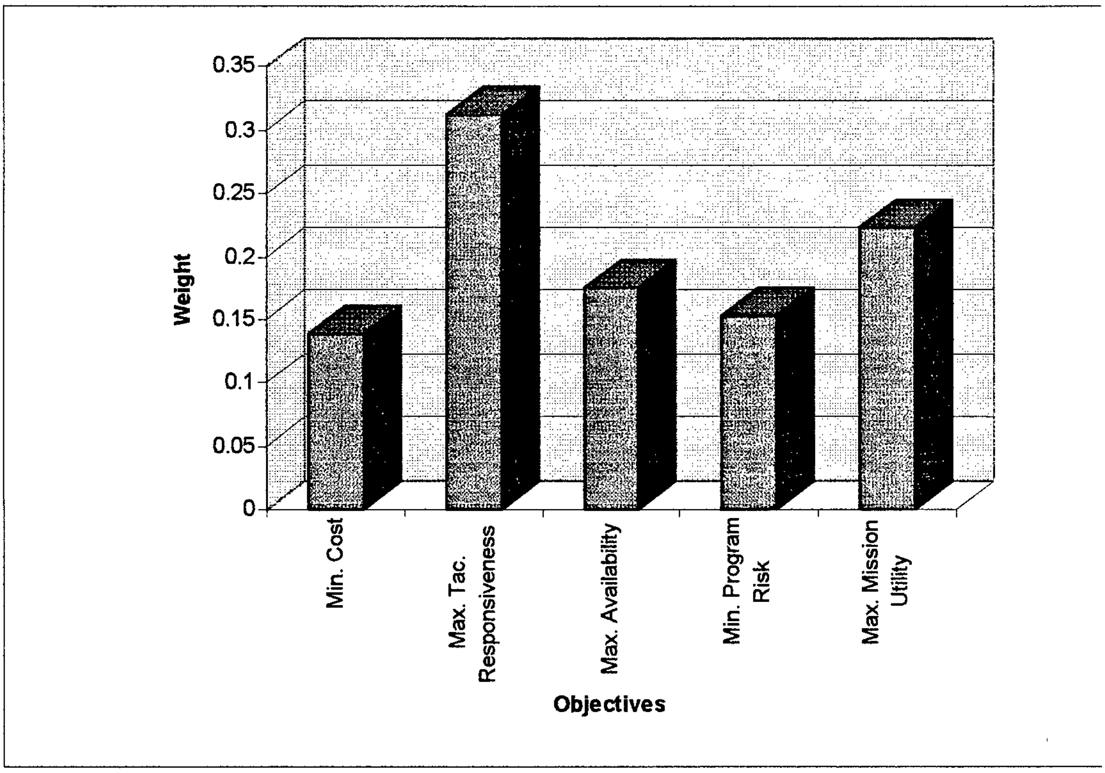


Figure 14-10: Top-level Objective Weights

The distribution of priorities for the tactical responsiveness objective is shown in Figure 14-11. The minimization of data latency has the highest priority. This is consistent with the recent emphasis of the military on combat theater information flow. As warfighters grow more and more dependent on satellite data, it is crucial that they receive satellite data as quickly as possible, with real-time data being the ideal standard. The weight of this objective demonstrates the importance of minimizing the time required to store, process, and down-link satellite data.

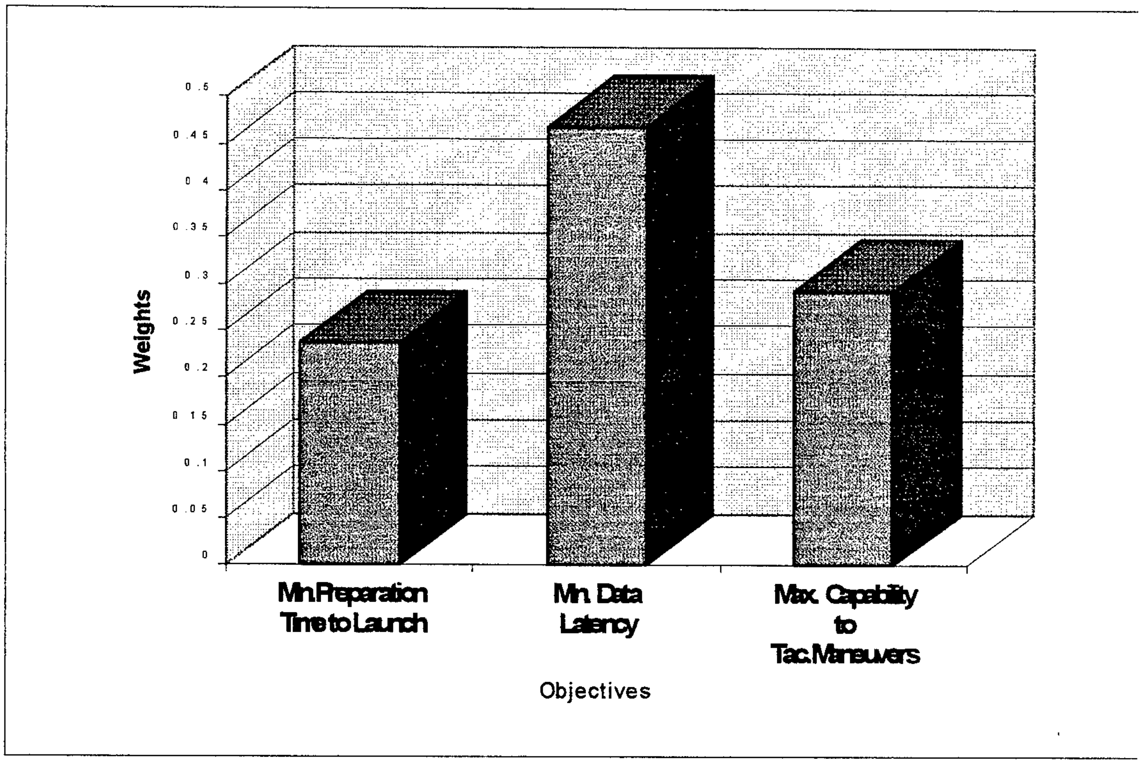


Figure 14-11: Tactical Responsiveness Weights

Figure 14-9 shows that reliability is the more important aspect of availability, while performance risk has the highest priority among the program risk objectives.

Figure 14-12 shows the weights of the ten sub-objectives under the mission utility objective. The minimization of pointing accuracy is the most important sub-objective, followed by the maximization of allowable mission module weight. Maximization of peak mission module power is the least important sub-objective.

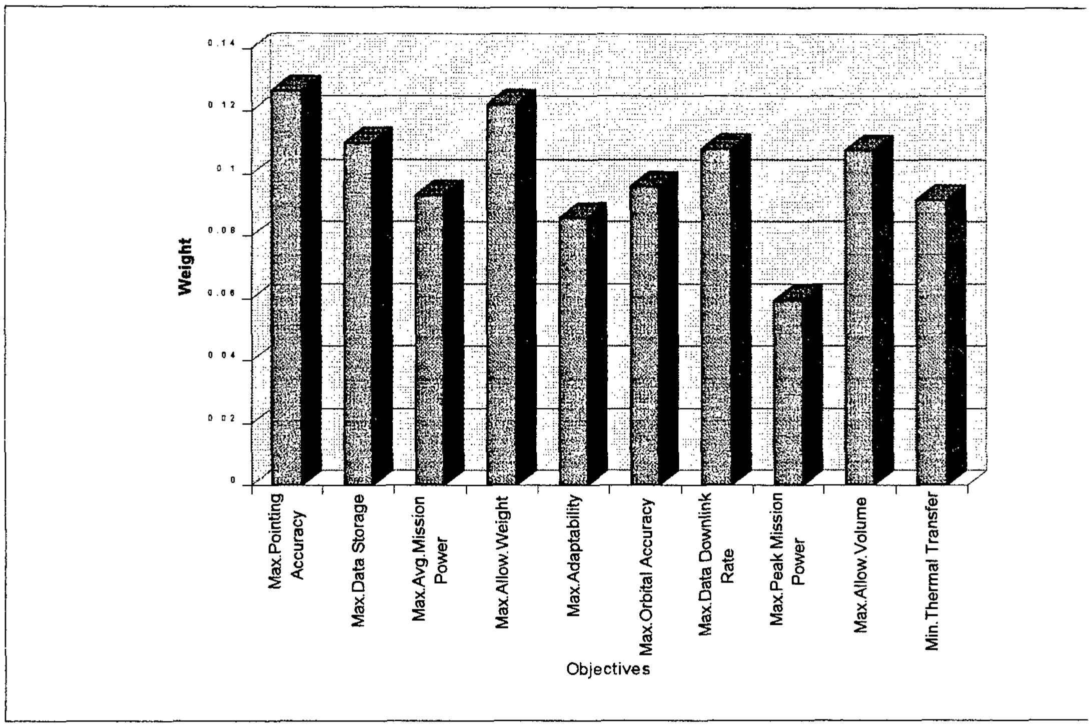


Figure 14-12: Mission Utility Weights

14.7 Scoring Function

The utility values and objective weights must be combined to form an overall utility function. This function yields an overall utility score for a given alternative solution, based on its performance of each objective and the relative importance of those objectives. Alternative systems solutions can be compared by their overall utility scores.

The utility function used for this study was additive, such that

$$U = k_1U_1 + k_2U_2 + \ldots + k_mU_m$$

where U is the overall utility, U_1, U_2, ... U_m are the individual utility functions for the m objectives, and k_1, k_2, ... k_m are the weights for each objective (Clemen, 1996:537). The use of an additive utility function must be justified, since it ignores any interaction among the attributes. For instance, some attributes may be complementary in nature, such that high achievement on all of them leads to more success than the sum of the individual successes (Clemen, 1996:576). In order to achieve accurate results with an additive utility function, additive independence must hold. Additive independence requires that the weights for the objectives at a given level in a branch of the hierarchy add up to one. According to Clemen, "When we are considering a choice among risky prospects involving multiple attributes, if additive independence holds, then we can compare the alternatives one attribute at a time" (Clemen, 1996:584).

Since this study was intended to produce concept-exploration design characteristics, and not detailed design recommendations, the team decided to avoid modeling the interaction among the objective attributes. According to Clemen,

> ... in extremely complicated situations with many attributes, the additive model may be a useful rough-cut approximation. It may turn out that considering the interactions among attributes is not critical to the decision at hand (Clemen, 1996:585).

Based on this reasoning, the additive utility function was deemed appropriate for this study.

14.8 Flexibility of the Value System

Since the value system drives all design efforts, changes in any of the elements of the value system can lead to different results. The objectives, weights, and utility functions used for this study could be modified upon further engineering efforts. It was not the intent of the team to create a rigid, unchanging value structure, but rather to create a robust framework within which intelligent decisions could be made. Now that the framework exists, it can be modified according to the changing desires of the CDM and the analysts who conduct further research on this topic.

15. Tradeoffs

15.1 System Level

Many important tradeoffs were analyzed at the system and subsystem level, and several critical design decisions were made. The purpose of this section of the report is to document these tradeoffs and decisions. Throughout the system study, the team has encountered many variables and options. Some of these have remained as variables, to be specified as elements of alternative system solutions for judgment against the value system criteria. However, many design decisions were made during this study, in order to narrow the solution space to a reasonable set of alternatives. Thus, the Modsat bus concept gained its shape throughout the study, as key decisions have built on each other. These decisions were made after performing tradeoffs between the alternatives, judging each against the Modsat objectives and constraints.

Much work was done at the subsystem level, and this is documented in section 15.5. This section discusses the system level tradeoffs.

15.1.1 One Satellite Per Launch Vehicle

Rather than attempt to design very small satellite busses for a multiple-payload launch package, the team chose to focus on one vehicle that will fill the payload bay of the chosen launch vehicle (LV). A key objective of the study is to allow for as much mission module capability as possible per bus. It would be inconsistent with this objective to force mission module designers to integrate with a microsat, or to make them spread their mission capability over a multiple-satellite configuration.

15.1.2 Launch Vehicle

The team originally chose the Pegasus XL, from Orbital Sciences Corporation (OSC), for reasons which are discussed in full in section 15.3. This LV is common for small low earth orbit (LEO) satellites. It is attractive for several reasons, among them being the ability to respond rapidly to mission needs and the flexibility of launch integration and location.

15.1.3 Basic Spacecraft Configuration

The basic configuration of a spacecraft depends on its primary means of attitude stabilization. Most LEO spacecraft are either two-axis stabilized via spinning the body, or three-axis stabilized via an internal device to control attitude.

Modsat requires three axes of pointing control, since it has articulated solar arrays (see section 15.5.6.4) and must accommodate nadir pointing mission modules. According to aerospace consultant Emery Reeves,

> "The spacecraft configuration must provide two axes of control for each item that is to be pointed. The spacecraft body has three axes so the body alone can satisfy one pointing requirement; for instance, one body axis (i.e., the yaw axis) can be pointed toward nadir by control about the other two axes (roll and pitch). It two items are to be pointed, then the spacecraft must be configured with at least one articulated joint between the two items. For illustration, a body mounted antenna can be pointed nadir by controlling two axes of body attitude. A solar array can then simultaneously be pointed toward the Sun by using the third body axis and providing a single solar array drive to control the solar array attitude

relative to the body...3-axis-controlled spacecraft generally use articulated panels" (Reeves, 1992:297).

Although two-axis systems are usually cheaper, lighter, and less complex than three-axis systems (Reeves, 1992:305), it was determined that the pointing requirements of both the solar arrays and the nadir pointing mission modules mandate the use of three-axis control. In the discussion on mission module requirements, it was stated that Modsat must provide pointing control at least as accurate as 0.1 degrees. According to Reeves, high accuracy pointing (<0.1 deg) implies the need for three-axis control.

The team also decided to use active control, with propulsion thrusters and reaction wheels (see section 15.5.1 and section 15.5.2). According to Reeves, "The active three-axis method gives us highly accurate pointing control, more efficient solar arrays (by allowing oriented planar arrays), and pointing of several payloads or spacecraft appendages" (Reeves, 1992:304).

15.1.4 Data Delivery Architecture

The users in the field would like to have their mission data as rapidly as possible. By requirement, Modsat must have S-band SGLS (space to ground-link subsystem) antennas which are compatible with the AFSCN. The SGLS link is limited to a maximum data rate of 1.024 Mbits/sec. Any tactical, warfighting satellite would benefit greatly from having its own high data rate downlink antenna(s), in addition to the SGLS antennas. Thus, the design team acknowledged that almost all Modsat satellites would require a high data rate antenna in addition to the required SGLS antennas. It was decided by the team to avoid selecting and integrating a particular type of antenna.

Rather, the mission planners will have the flexibility to choose antennas that suit their needs. Thus, all Modsat bus designs accounted for the placement of a high data rate package, whether in the bus itself (as a standardized add-on) or as part of the mission module.

15.1.5 Adaptability and Modularity

As mentioned in the Problem Definition section, the CDM desires Modsat to have a flexible, adaptable design, such that additional capability can inserted, or excess capability can be removed, depending on the needs of each mission module. There have been other programs with similar design goals. One such program was the Advanced Technology Standardized Satellite Bus (ATSSB), under the direction of ARPA. An interview was conducted with the former director of the ATSSB program, Colonel (Retired) Ed Nicastri. Mr. Nicastri explained that his program examined this issue of the adaptable configuration. They chose a modular approach, whereby components are plugged in to the bus like circuit cards, and removed when not needed.

Nicastri advised that it is far better to design in a high degree of capability, and remove the excess, than to start with the minimum and add as necessary. The latter approach implies a skeletal structure, where components are attached as required for each mission. Nicastri's program determined that this configuration would be a nightmare for integration and configuration control, and would be far too expensive. Thus, it is better to start with the maximum capability and remove from there (Nicastri, 1996).

The approach chosen by the team takes ARPA's experience into consideration. A modular approach will be used, whereby all busses are manufactured with all foreseeable component options attached via removable interfaces. Of course, center of mass and attitude control constraints may preclude the removal of some excess components, depending on the characteristics of the mission module in question.

15.1.6 Maximum Capability

The Modsat bus must accommodate a wide variety of mission modules, and must therefore cater to the most demanding customers. This implies that there will be excess capability on many missions. Some of this may be mitigated by the modular architecture discussed above, but certainly not all excess capability can be removed. The notion of built-in excess that will accompany the mandated, standardized interface is a feature that may encounter resistance from industry. According to Richard Warner, Chief Technical Officer of AeroAstro Corporation, "satellite designers crave efficiently engineered, customized packages" (Warner, 1996). However, given the objectives of Modsat, this cannot be avoided.

15.1.7 On-Board Propulsion

As mentioned in section 15.1.3, the team decided to use active three-axis control, with propulsion thrusters. The alternative method of three-axis control is passive control, which uses gravity gradient techniques or magnetic torquers to control and slew the spacecraft. According to Reeves, "Passive techniques can provide coarse control to support low-accuracy pointing requirements and simple spacecraft" (Reeves, 1992:304).

Magnetic torquers are often used in place of the bulkier propellant-thruster configurations to reduce spacecraft momentum. However, magnetic torquers cannot perform ΔV maneuvers (Everett, 1996). In section 15.4, it will be shown that Modsat requires ΔV capability to maintain its altitude at LEO. Moreover, thrusters are required to perform tactical repositioning ΔV maneuvers, which may very well be required as part of Modsat's tactical mission profile (see Problem Definition).

15.1.8 Data Processing

Most space sensor applications require some degree of processing of the mission data, prior to its transmission to the ground. In any given custom-built satellite, it is possible for such data to be handled by the main spacecraft processor, along with all of its other processing functions. However, each mission type has its own very unique processing requirements, and it would be impossible to satisfy all with one processor. In fact, many instruments have their own mini-computers. Thus, the team decided that mission data processing must be performed by the mission module. However, data storage will be available in the bus processor, as stated in the requirements.

15.1.9 The Bus-To-Mission Module Interface

Another key part of this study is the specification of a standardized interface between the bus and all mission modules. This very issue has been addressed by the Aerospace Corporation, and the results are documented in "An Approach To Rapid Payload Integration: The Spacecraft-To-Payload Interface Guideline (SPIG), Version 1"

(Aerospace Corporation, 1996). According to this document, "the SPIG is intended to serve as a core building block on which the payload-to-spacecraft interface can be designed."

In the production of the SPIG, much systems engineering work has been accomplished to suggest an optimum standardized interface. The team decided to use the SPIG interface in lieu of designing their own interface.

15.2 Reliability Analysis

15.2.1 Overview

A major tradeoff in the design of any new system is the trade between the reliability and cost of a system. Due to the generally high cost of past and present space efforts, this takes on an even greater level of importance. In addition, the current emphasis on finding ways to do the space mission "better, faster, cheaper" adds further importance to this trade. Finding the optimum level of reliability to meet cost and performance goals will help narrow the solution space of the system design. Thus, as part of the overall system design/analysis process, a cost vs. reliability trade study was performed to determine the starting point from which a system-level reliability approach should proceed.

The formal definition of "reliability" adhered to in this design project is "the probability that the system will perform its intended function for a specified interval of time under stated conditions" (Ramakumar, 1993:3). The "stated conditions" assumed in this project are the natural (non-man-made) hazards common to the low-Earth orbit space environment, including thermal, vacuum, and radiation hazards. For the purposes of this study, it is assumed that the spacecraft has survived the launch and has been successfully inserted into its operational orbit. Thus, mission failures attributed to the launch vehicle (including the kick motor, if present) were not considered. This does not mean that failures induced by launch stresses were not included. Rather, it means that the launch vehicle successfully placed the satellite in it's proper orbit.

As part of the cost vs. reliability trade study, a MATLAB model was created that allows the user to determine the overall system reliability of the spacecraft based on Mean Time Between Failures (MTBF) and redundancy levels for individual subsystems. The model is explained in detail in Volume III of the report.

15.2.2 Reliability Strategies

As previously stated, the objective of the cost vs. reliability trade study was to determine the optimum starting point for selecting the reliability approach to be used in designing the spacecraft. Two approaches were studied: fault avoidance and fault tolerance. "Fault avoidance" seeks to prevent failure by tactics such as providing generous design margins, using high-quality parts, and thorough inspection and testing of spacecraft components, subsystems, and the assembled spacecraft (Hecht, 1993:700). In contrast, "fault tolerance" seeks not to fervently avoid failure, but to allow for continued spacecraft operation despite the failure, generally through either equipment redundancy (i.e. spares) or functional redundancy (other parts of the spacecraft not optimized for a particular function assuming the duties of the failed equipment). Equipment and functional redundancy can be implemented at the subsystem level, component level, or within a spacecraft component (e.g. an extra gyroscope within an inertial measurement unit)

(Hecht, 1992:701). Within the context of this study, fault avoidance was limited to using high-quality parts, and fault tolerance was limited to equipment redundancy.

15.2.3 Methodology

In considering fault avoidance vs. cost, three classes of components were studied: commercial-grade, Class B, and Class S. *Commercial-grade* components are similar to what can be purchased at a local electronics store or from a mail-order catalog. An example would be purchasing a Pentium processor from the local computer store and using it as the central processing unit in the spacecraft's Command and Data Handling subsystem. Such hardware is generally inexpensive and readily available, but lack the extensive testing, quality-control, and documentation of other component classes (Hecht, 1996:67, 76) *Class B* components can be thought of as components that have been built and tested to meet general military specifications and standards (MIL-SPEC/STD) (Muolo, 1993:315). Because of the often rugged environments military hardware must operate in, they tend to be built to more exacting specifications, such as government-approved materials, suppliers, manufacturing processes, packaging, and transport (Hecht, 1996:67). Class B component must endure much more thorough and rigorous testing and must often be accompanied by extensive documentation tracing their development history, something not required of commercial-grade components (Hecht, 1992:700). *Class S*, or "space-qualified" components are considered to be at the top of the hierarchy when it comes to component quality. In addition to the requirements imposed on Class B components, Class S components are subjected to additional, space environment-unique survivability tests and are always accompanied by an extensive documentation trail tracing their history (Hecht, 1996:68). Accordingly, the costs of components increase as one moves up the quality hierarchy.

Fault tolerance vs. cost was examined by comparing spacecraft cost vs. system redundancy level. Three classes of redundancy were used: single-string, double-string, and triple-string. Single string means that only the subsystems essential to an operational spacecraft are included, and none of the subsystems are duplicated. Double string means that all essential subsystems have one primary and one standby that can be brought on-line by activating a switch. For the purposes of this study, the switching mechanism is assumed to have perfect reliability. In reality, this is not the case. Despite this, as long as this assumption is consistent throughout the study, it should have no effect on the outcome, given the study's comparative nature. The concept behind triple string spacecraft is identical to double string, except each primary subsystem has two standbys.

A third approach in improving spacecraft reliability was not included in the study but is nonetheless worthy of mention. It takes a holistic approach to reliability optimization, focusing less on individual components and subsystems and more on the "expected" reliability of the system as a whole. Enhancing expected reliability differs (albeit in a subtle manner) from traditional "predicted" reliability improvement (the goal of fault tolerance and fault avoidance) in that it incorporates measures such as providing extra protection to critical components from the space environment, operating components and subsystems well within their design limits (i.e. de-rating operational capability), incorporating generous power generation capacity to allow graceful degradation on orbit, rigorous system-level testing, and other system-level measures. Expected reliability can be thought of as an "update" to the original predicted reliability.

Nowadays, designers and builders, particularly of small, short-mission satellites, are finding techniques that focus on the overall system-level as a more accurate and cost-effective means of predicting and improving reliability (Everett, 1996; Nicastri, 1996; Jackson, 1996; Warner, 1996). The following concept map illustrates the complexities of the cost-reliability relationship.

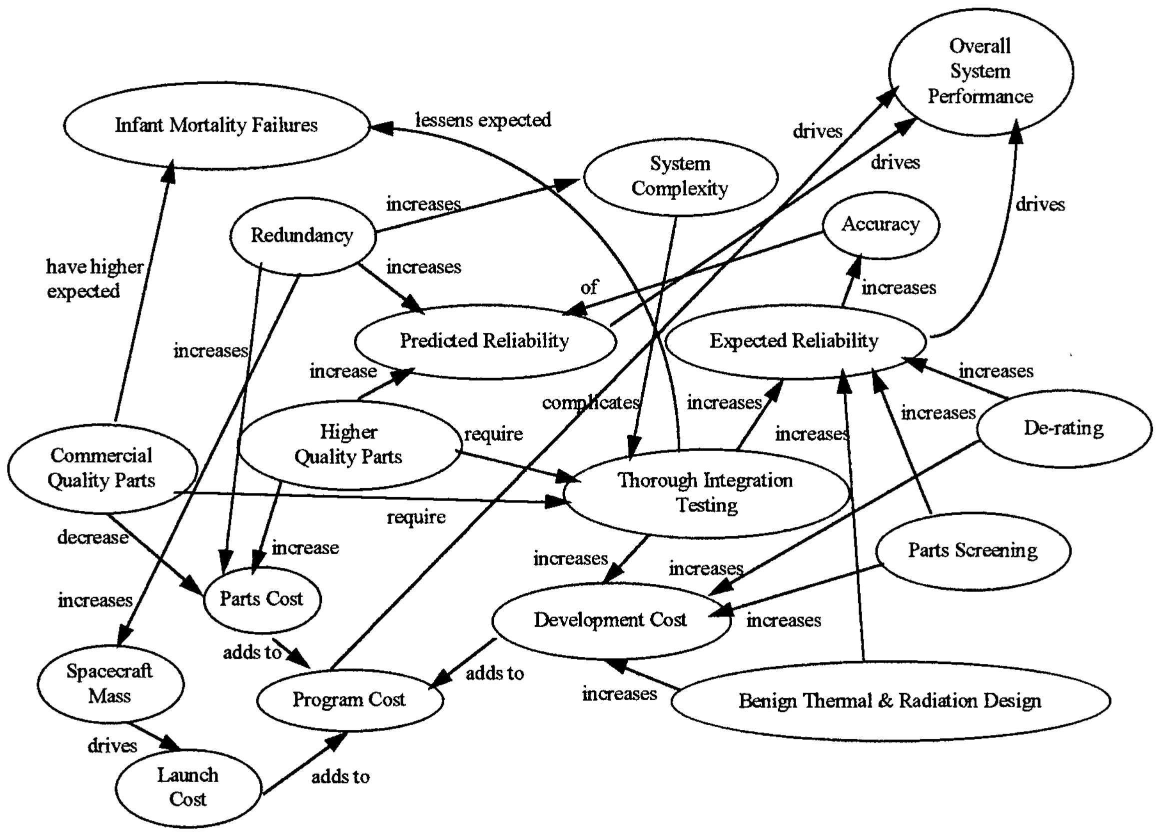


Figure 15-1: Cost-Reliability Concept Map

The data used in this study comes primarily from a study of on-orbit spacecraft failures from the early 1960s to 1984, published by Rome Air Development Center, Air Force Systems Command. The study examined failures in over 300 satellites from 96 programs (Hecht, 1985:1). Failures are broken out by cause, severity, affected subsystem, failure date/time, and other parameters (Hecht, 1985:7). Throughout the study, various combinations of failures are examined and patterns are noted where evident.

15.2.4 Assumptions

Data was extracted from the report using the following assumptions (Hecht, 1985:62-109):

1. Only consider post-1977 failure data (allows for technology differences pre- and post-1977)
2. Mission length before failure does not exceed one year
3. Only the following classes of failures were considered relevant:

   Class 1 (Critical Failure -- Entire satellite or a major mission function fails)

   Class 2 (Single Point Failure -- Major assembly or component failure)

   Class 5 (Degraded Performance -- Degraded performance of a mission function)

In addition, the following assumptions were also applied:

4. Components and subsystems in the RADC report are assumed to be Class B (This seems logical for two reasons: a) Class S parts were not available until after the early 1980s (Hecht, 1996:295); b) the idea of using ordinary commercial grade, non-MILSPEC/STD or Class S components for space applications is a relatively new phenomenon).
5. The failure rates of Class S components are approximately 25% of Class B components and 10% of high-grade commercial components (Hecht, 1992:700).
6. The cost of Class S components is roughly four times the cost of Class B components, and roughly ten times the cost of commercial-grade components (this assumption was examined in the sensitivity analysis).

7. The recurring (i.e. production) cost of the spacecraft is approximately \$30M (derived from the Cost Modeling section in Vol. III of the thesis).

15.2.5 Analysis

Given these assumptions, the overall satellite failure rate under the stated assumptions was determined by the following procedure:

Total # of failures (pre- and post-1977) = 2346 (Hecht, 1985:96-97)

Total # of flights (pre- and post-1977) = 321 (Hecht, 1985:96-97)

Overall failure rate = 2346/321 = 7.308 failures/spacecraft-year

Since the post-1977 overall failure rate is roughly twice the pre-1977 overall failure rate (Hecht, 1985:49), pre-1977 failures should account for one-third of the total overall failure rate, and post-1977 failures should account for the remaining two-thirds. Hence,

Pre-1977 failure rate = 1/3 x 7.308 = 2.436 failures/spacecraft-year

Post-1977 failure rate = 2/3 x 7.308 = 4.872 failures/spacecraft-year

# of failure due to bus (pre- and post-1977) = 1832 (Hecht, 1985:73)

Total # of failures = 2469 (considered valid despite slight (~5%) discrepancy with above number) (Hecht, 1985:73)

The percentage of failures attributable to the bus was 74.2%. (Hecht, 1985:73) Hence,

Computed failure rate for bus, post-1977 = 4.872 (0.742) = 3.615 failures/spacecraft-year

The fraction of Class 1,2, and 5 failures were given as follows: (Hecht, 1985:52)

% of failures that were Class 1 = 3%

% of failures that were Class 2 = 5%

% of failures that were Class 5 = 20%

Therefore,

Failure rate (bus, post-1977, Class 1,2, and 5 failures) = (0.03+0.05+0.20) x 3.615 = 1.0122 failures per spacecraft-year

The fraction of failures during the first year of on-orbit operations as broken out according to subsystem is as follows: (Hecht, 1985:75)

Telemetry, Tracking, and Commanding (TTC) = 0.400

Guidance and Control (ADCS) = 0.133

Power (EPDS) = 0.150

Data Handling (CDH) = 0.200

Thermal = 0.067

Propulsion = 0.050

Structure = 0

Thus, for a one-year mission under the previously stated assumptions, the failure rates are as follows (let λ represent failure rate):

λ_{TTC} = 0.400 (1.0122) = 0.40488 failures/spacecraft-year

λ_{ADCS} = 0.133 (1.0122) = 0.13462 failures/spacecraft-year

λ_{EPDS} = 0.150 (1.0122) = 0.15183 failures/spacecraft-year

λ_{CDH} = 0.200 (1.0122) = 0.20244 failures/spacecraft-year

$\lambda_{Thermal}$ = 0.067 (1.0122) = 0.06782 failures/spacecraft-year

$\lambda_{Propulsion}$ = 0.050 (1.0122) = 0.05061 failures/spacecraft-year

$\lambda_{Structure}$ = 0 (1.0122) = 0 failures/spacecraft-year

In order to account for the advancement of technology since 1984 (and a corresponding increase in component reliability), the above failure rates were divided by an "improvement factor". The improvement factor was calculated by taking λ_{CDH} and dividing it by the failure rate of a modern Class B CDH subsystem (Berget and Turner, 1992:389) :

$$\text{Improvement factor} = \lambda_{CDH} / \text{modern CDH failure rate}$$

$$= 0.20244/0.05462 \approx 3.7$$

An assumption implicit in the use of this factor is that all subsystems have improved by a factor of 3.7 since 1984. Dividing the failure rates of all of the subsystems by 3.7,

$\lambda_{TTC} = 0.109427$ failures/spacecraft $\lambda_{Thermal} = 0.01833$ failures/spacecraft

$\lambda_{ADCS} = 0.036384$ failures/spacecraft $\lambda_{Propulsion} = 0.013678$ failures/spacecraft

$\lambda_{EPDS} = 0.041035$ failures/spacecraft $\lambda_{Structure} = 0$ failures/spacecraft

$\lambda_{CDH} = 0.054714$ failures/spacecraft

In order to compute subsystem reliability and determine the effects of standby redundancy, the following formulas were used (Ramakumar, 1993:196) :

$R(t) = e^{-\lambda t}$ No redundancy (i.e. single-string)

$R(t) = e^{-\lambda t} [\ 1+\lambda t\ ]$ Single redundant unit (i.e. double-string)

$R(t) = e^{-\lambda t} [\ 1+ \lambda t + (\lambda t)^2/2!\ ]$ Two redundant units (i.e. triple-string)

where $R(t)$ = reliability

t = time ($t = 1$, one year)

λ = failure rate (failures per spacecraft over the 1- year time period)

The results are listed in Table 15-2, Table 15-3, and Table 15-4.

Table 15-2: Reliability and Failure Probability for a 1-year Mission;

Single-string Spacecraft

	Reliability			Failure Probability		
Subsystem	Class B	Class S	Commercial	Class B	Class S	Commercial
TT&C	0.8963476	0.9730141	0.760661	0.103652	0.026986	0.2393391
ADCS	0.9642702	0.9909453	0.913055	0.03573	0.009055	0.0869453
EPDS	0.9597954	0.9897937	0.902499	0.040205	0.010206	0.0975011
CDH	0.9467563	0.9864147	0.872159	0.053244	0.013585	0.1278412
Thermal	0.9818372	0.9954281	0.95521	0.018163	0.004572	0.0447902
Propulsion	0.9864147	0.9965862	0.966382	0.013585	0.003414	0.0336179
Structure	1	1	1	0	0	0
SYSTEM	0.7606609	0.9338944	0.504635	0.239339	0.066106	0.4953645

Table 15-3: Reliability and Failure Probability for a 1-year Mission;

Double-string Spacecraft

	Reliability			Failure Probability		
Subsystem	Class B	Class S	Commercial	Class B	Class S	Commercial
TT&C	0.9944322	0.9996326	0.968753	0.005568	0.000367	0.0312469
ADCS	0.9993539	0.9999589	0.996106	0.000646	4.11E-05	0.0038943
EPDS	0.9991807	0.9999477	0.995084	0.000819	5.23E-05	0.0049157
CDH	0.9985567	0.9999073	0.991456	0.001443	9.27E-05	0.008544
Thermal	0.999834	0.9999895	0.998982	0.000166	1.05E-05	0.0010184
Propulsion	0.9999073	0.9999942	0.999428	9.27E-05	5.83E-06	0.0005715
Structure	1	1	1	0	0	0
SYSTEM	0.991286	0.9994303	0.950519	0.008714	0.00057	0.0494805

Table 15-4: Reliability and Failure Probability for a 1-year Mission; Triple-string Spacecraft

	Reliability			Failure Probability		
Subsystem	Class B	Class S	Commercial	Class B	Class S	Commercial
TT&C	0.9997988	0.9999967	0.997217	0.000201	3.34E-06	0.0027833
ADCS	0.9999922	0.9999999	0.999883	7.81E-06	1.25E-07	0.0001172
EPDS	0.9999888	0.9999998	0.999833	1.12E-05	1.79E-07	0.0001667
CDH	0.9999738	0.9999996	0.999615	2.62E-05	4.22E-07	0.0003851
Thermal	0.999999	1	0.999985	1.01E-06	1.6E-08	1.55E-05
Propulsion	0.9999996	1	0.999994	4.22E-07	6.65E-09	6.496E-06
Structure	1	1	1	0	0	0
SYSTEM	0.9997522	0.9999959	0.996528	0.000248	4.09E-06	0.0034721

Table 15-5 combines the system reliability data in the tables above with the appropriate cost figures, using a $30M class S spacecraft as the baseline:

Table 15-5: Spacecraft Cost vs. Reliability (1st Cost Assumption)

Class	R(t=1yr)	Cost ($M)
Commercial, 1X	0.504635	3
Commercial, 2X	0.950519	6
B, 1X	0.760661	7.5
Commercial, 3X	0.996528	9
B, 2X	0.991286	15
B, 3X	0.999752	22.5
S, 1X	0.933894	30
S, 2X	0.99943	60
S, 3X	0.999996	90

Table 15-6 shows the results of changing the cost assumption (Assumption # 6) so that the cost ratio from commercial class to Class B (5:1) is the same as going from Class B to Class S. A $30M Class S spacecraft is still the baseline.

Table 15-6: Spacecraft Cost vs. Reliability (2nd Cost Assumption)

Class	R(t=1yr)	Cost ($M)
Commercial, 1X	0.504638	1.2
Commercial, 2X	0.950519	2.4
Commercial, 3X	0.996528	3.6
B, 1X	0.760661	6
B, 2X	0.991286	12
B, 3X	0.999752	18
S, 1X	0.933894	30
S, 2X	0.99943	60
S, 3X	0.999996	90

Table 15-7 shows the results of changing the cost assumption again so that the cost of Class S is 300% of commercial and Class B is 200% of commercial, with a $30M Class S spacecraft as a baseline.

Table 15-7: Spacecraft Cost vs. Reliability (3rd Cost Assumption)

Class	R(t=1yr)	Cost ($M)
Commercial, 1X	0.50464	10
Class B, 1X	0.76066	20
Commercial, 2X	0.95052	20
Class S, 1X	0.93389	30
Commercial, 3X	0.99653	30
Class B, 2X	0.99129	40
Class S, 2X	0.99943	60
Class B, 3X	0.99975	60
Class S, 3X	1	90

Graphical depictions of the data are in Figure 15-2, Figure 15-3, and Figure 15-4:

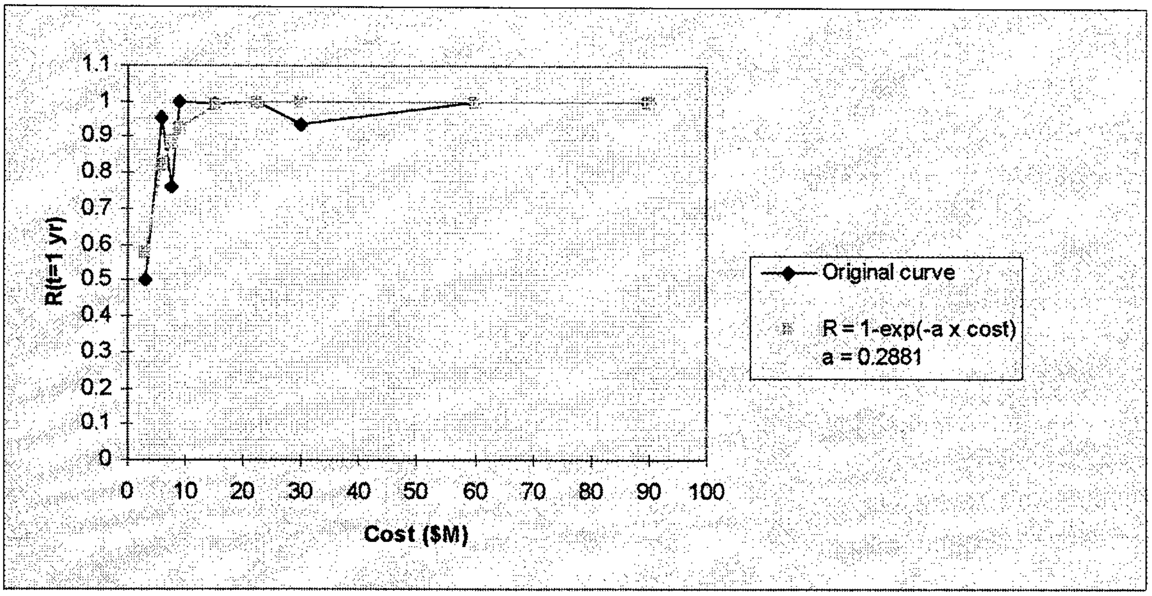


Figure 15-2: Cost vs. Spacecraft Reliability Improvement Under 1st Cost Assumption

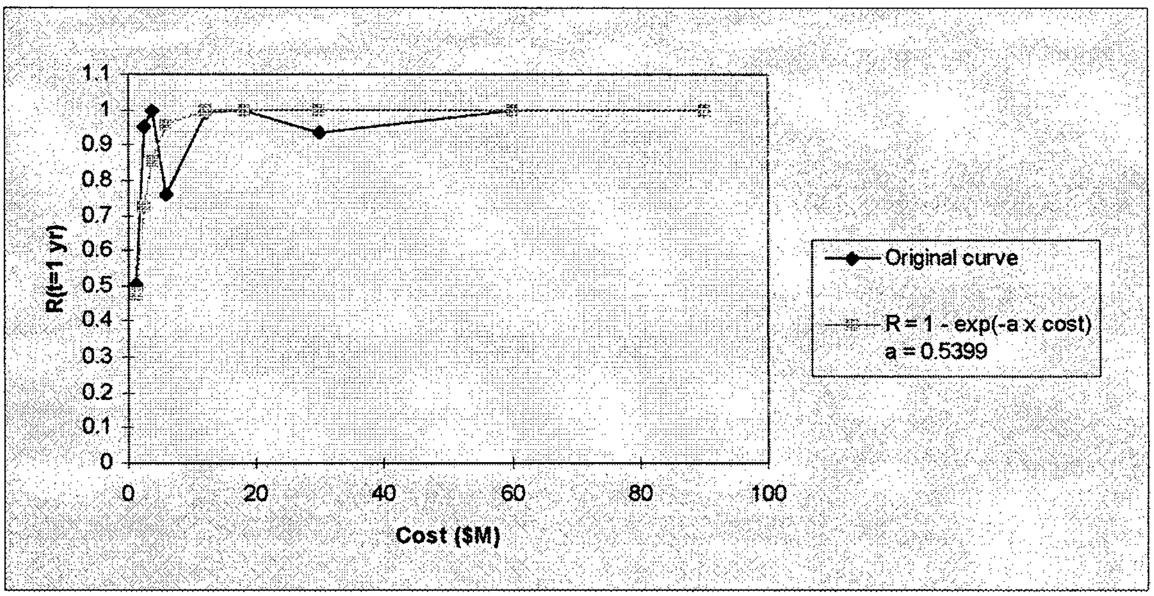


Figure 15-3: Cost vs. Spacecraft Reliability Improvement Under 2nd Cost Assumption

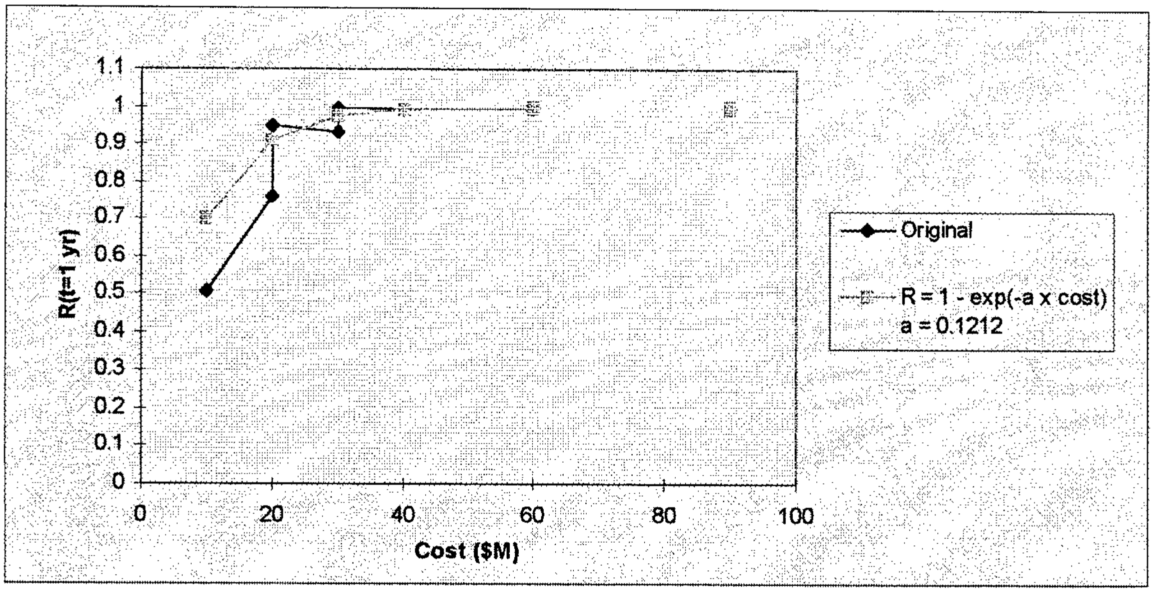


Figure 15-4: Cost vs. Spacecraft Reliability Under 3rd Cost Assumption

An exponential curve has been fitted to the data in all three cases, superimposed over the original data. It can be seen that the optimum choices between cost and reliability (i.e. the "knee" of the fitted curve) seems to be a single-string, Class B spacecraft, or a double or triple-string spacecraft with commercial components.

Table 15-8: Cost and Reliability for Spacecraft with Various Class/Redundancy Levels

Cost Assumption #1								
Cost($M)	R(t=1yr)	Class	Cost($M)	R(t=1yr)	Class	Cost($M)	R(t=1yr)	Class
3	0.50464	comm, 1X	7.5	0.76066	Class B, 1X	30	0.933894	Class S, 1X
6	0.95052	comm, 2X	15	0.99129	Class B, 2X	60	0.99943	Class S, 2X
9	0.99653	comm, 3X	22.5	0.99975	Class B, 3X	90	0.999996	Class S, 3X
Cost Assumption #2								
Cost($M)	R(t=1yr)	Class	Cost($M)	R(t=1yr)	Class	Cost($M)	R(t=1yr)	Class
1.2	0.50464	comm, 1X	6	0.76066	Class B, 1X	30	0.933894	Class S, 1X
2.4	0.95052	comm, 2X	12	0.99129	Class B, 2X	60	0.99943	Class S, 2X
3.6	0.99653	comm, 3X	18	0.99975	Class B, 3X	90	0.999996	Class S, 3X
Cost Assumption #3								
Cost($M)	R(t=1yr)	Class	Cost($M)	R(t=1yr)	Class	Cost($M)	R(t=1yr)	Class
10	0.50464	comm, 1X	20	0.76066	Class B, 1X	30	0.933894	Class S, 1X
20	0.95052	comm, 2X	40	0.99129	Class B, 2X	60	0.99943	Class S, 2X
30	0.99653	comm, 3X	60	0.99975	Class B, 3X	90	0.999996	Class S, 3X

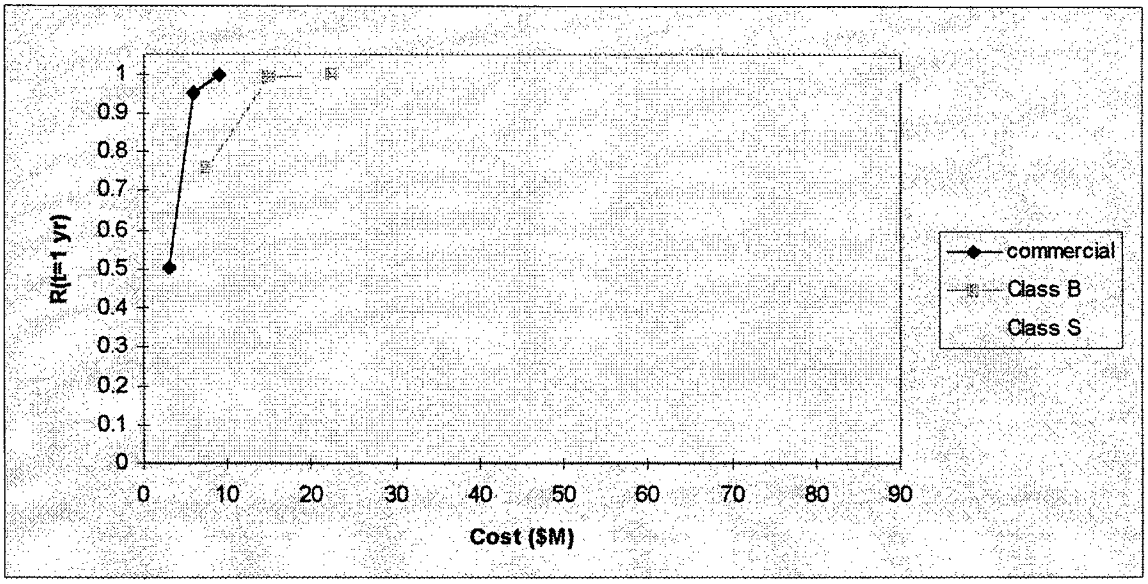


Figure 15-5: Cost vs. Reliability for Different Component Classes (1st Cost Assumption)

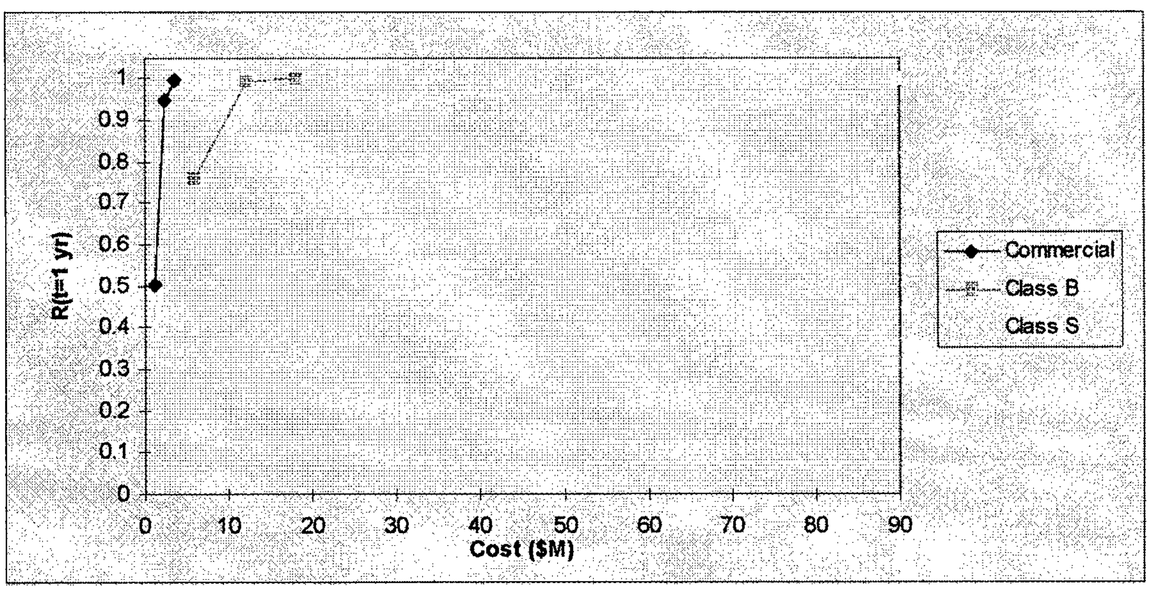


Figure 15-6: Cost vs. Reliability for Different Component Classes (2nd Cost Assumption)

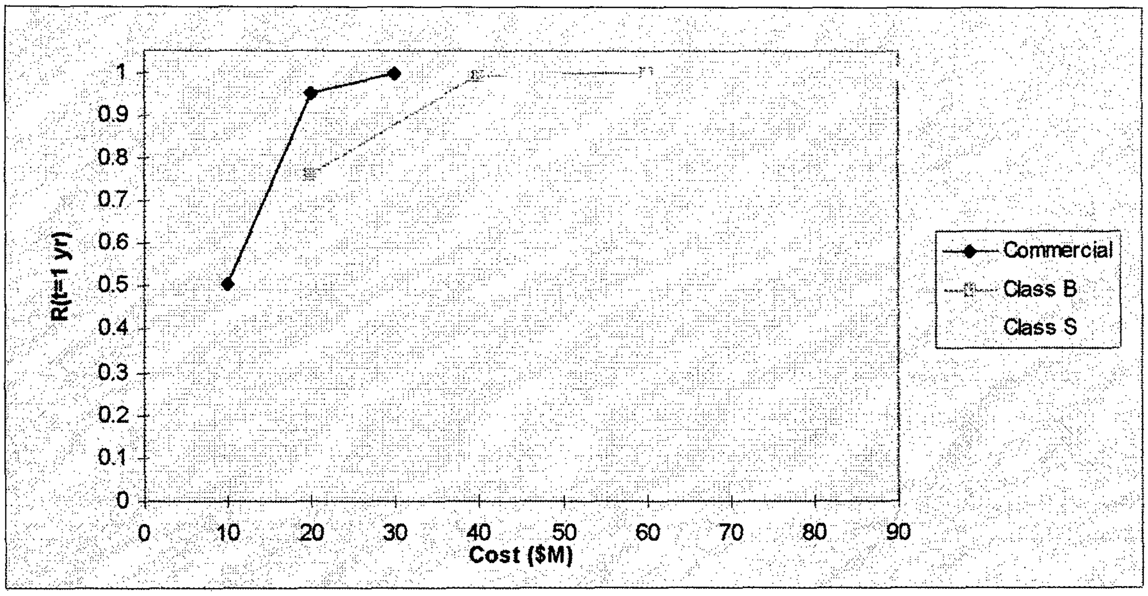


Figure 15-7: Cost vs. Reliability for Different Component Classes (3rd Cost Assumption)

Figure 15-5, Figure 15-6, and Figure 15-7 show a marked increase in reliability in going from single-string Class B to redundant commercial-grade components for about the same cost. However, it is vital to note that the assumption about the increase in costs between redundancy levels do not take into account the additional costs associated with redundant systems such as added launch costs (due to extra weight), the costs of testing and integrating redundant components, and other "hidden" costs. If these costs were to be added, the cost-reliability advantage of commercial-class with redundancy would be reduced.

15.2.6 Conclusion

Given today's push towards small, inexpensive spacecraft, the spacecraft designer can no longer afford to maximize reliability without regard to cost. The sky is no longer the limit insofar as performance requirements are concerned, and in the context of reliability, desired reliability must be balanced with costs. A detailed study of past spacecraft failures combined with some simple assumptions on cost showed that a commercial-class with redundancy would be a good starting point in optimizing reliability and cost. Of course, depending on program cost and performance goals and the availability of Class B and Class S components, the designer should tailor the cost-reliability balance to mission needs. Finally, it is likely that an even better balance can be achieved by designing the spacecraft using high-reliability components and multiple-string schemes into those subsystems that would benefit the most (TT&C, CDH), instead of an all-or-nothing approach.

15.3 Launch Vehicle Considerations

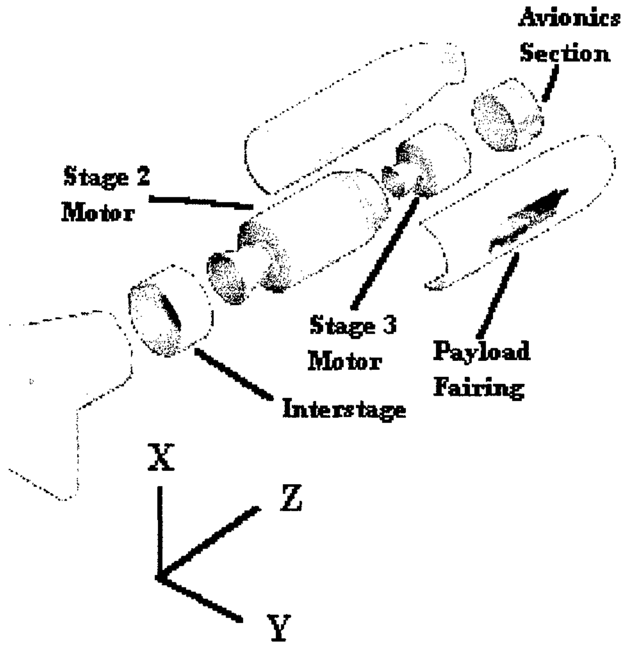
Although launch vehicle (LV) considerations are in the external environment of the engineering of the spacecraft bus, the LV payload constraints have a great influence on the design of the bus. Thus, even though the design and configuration of a launch vehicle was not a goal of the design team, it was necessary to choose a launch vehicle configuration . This choice placed constraints on the design of the alternatives. The sponsor's guidance required the satellite bus to be within the Pegasus and Lockheed Martin Launch Vehicle (LMLV) weight class (Rooney, 1996). Therefore, the team decided to choose one of these launch vehicles.

The selection of a launch vehicle for this project could have become a complex and intricate study within itself. Given the limited time and resources of this study, the team decided to select a LV that met the tactical needs of the sponsor and provided enough energy to put the desired spacecraft weight class into orbit. The Pegasus XL was chosen as the baseline LV. This decision was based on the success and experience of Orbital Science Corporation's Pegasus LVs. The air-launched nature of the Pegasus vehicles offers a tactical advantage over other conventional ground-based LVs. The capability of being rapidly deployed and launched into any inclined Low Earth Orbits (LEO) from any latitude and longitude is another distinct advantage of the Pegasus. More specifically, the Pegasus XL was chosen over the standard Pegasus vehicle because the XL version offers more mass-to-orbit performance. Other small satellite programs have also selected the Pegasus XL as the LV of choice (Wertz, 1996:350). One such program is NASA's Small Explorer (SMEX) program, which provides standardized buses for observation missions (Small Explorer, 1996:WWWeb).

It should be noted that the launch vehicle of choice can be changed in future studies. However, the choice of LV strongly influences the design of the bus. Thus, if the LV is changed to a booster other than the Pegasus XL, spacecraft bus designs may differ from those produced in this study.

15.3.1 Pegasus XL

The Pegasus XL is a winged, three-stage, solid rocket booster that weighs approximately 22,680 kg (50,000 lbm) and measures 16.9 m (55.4 ft) in length and 1.27 m (50 in) in diameter. An expanded view of the vehicle is shown in Figure 15-8. The rocket is lifted by a carrier aircraft, usually a Lockheed L-1011, to a level flight condition about 11,580 m (38.000 ft) and Mach 0.79 before it is released for launch (Orbital Science Corporation, 1993:2-1).



Source: OSC, 1993:2-2

Figure 15-8: Pegasus XL Expanded View

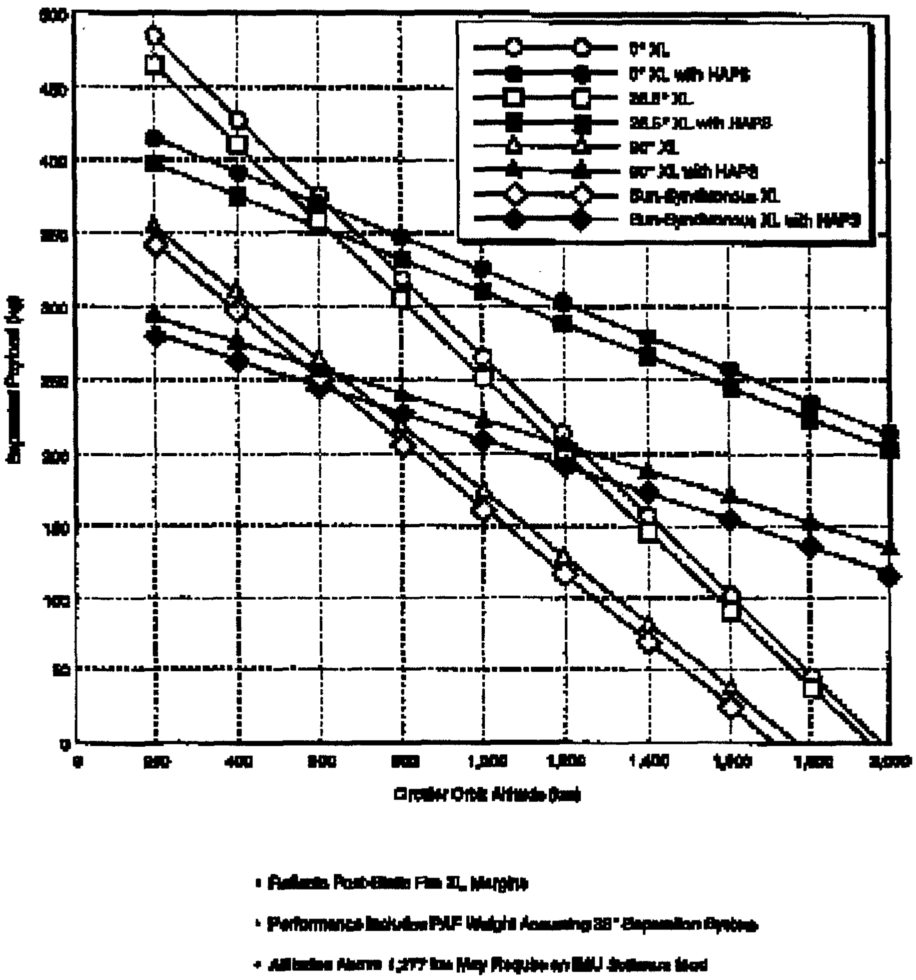
There are two major Pegasus XL features that impose restrictions on potential satellite bus designs. One is the Pegasus XL mass-to-orbit performance capability; the other is the booster's payload fairing dimensions.

15.3.2 Mass-to-Orbit Performance

An important characteristic of any launch vehicle is the amount of mass it can place into orbit. The mass that the Pegasus XL can deliver depends on the altitude and inclination of the desired orbit, and whether the booster uses an optional Hydrazine Auxiliary Propulsion System (HAPS).

The purpose of the HAPS is twofold. It improves orbital injection accuracy and increases mass-to-orbit capability for satellites placed into altitudes greater than approximately 600 kilometers. The HAPS contains a centrally mounted tank loaded with

72 kilograms of hydrazine, a highly pressurized regulated gas source, and three fixed, axially directed thrusters (OSC, 1993:2-6). The mass-to-orbit performance capability for the Pegasus XL is provided in Figure 15-9.

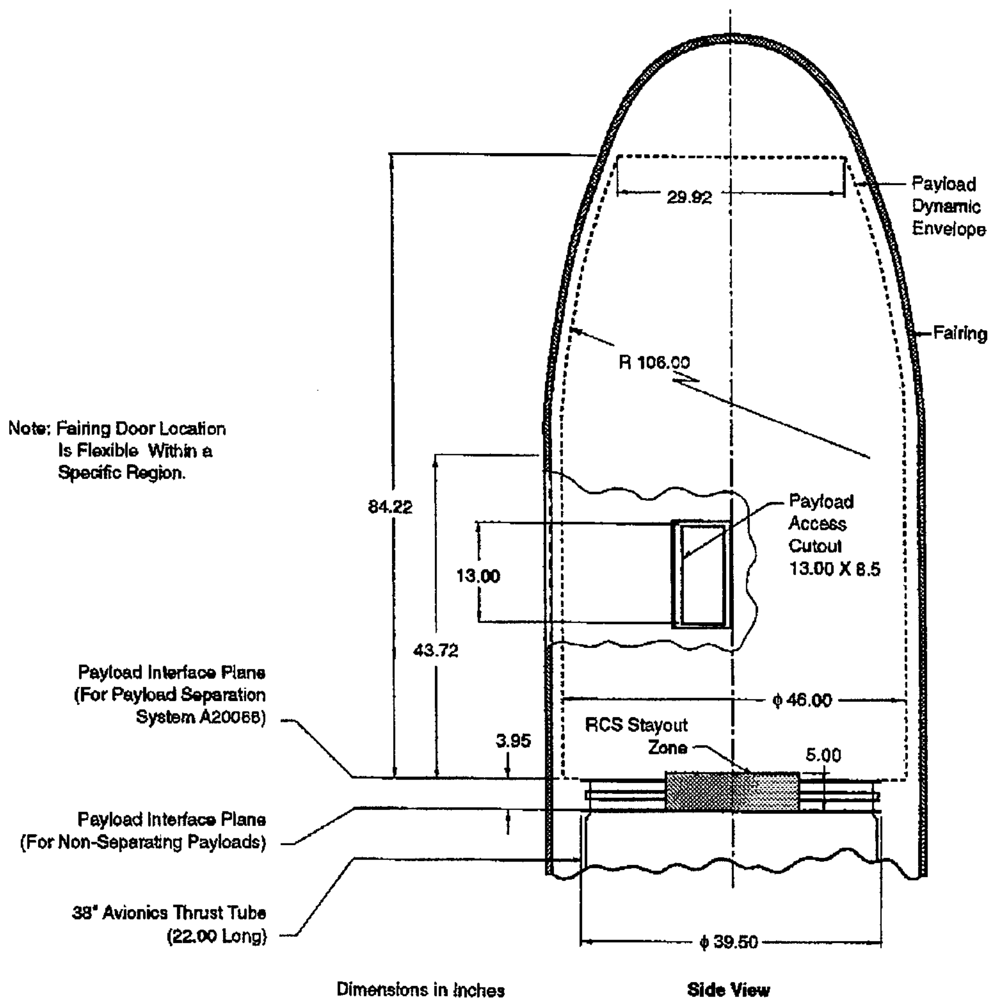


Source: OSC, 1993:3-3

Figure 15-9: Pegasus XL Performance Capability

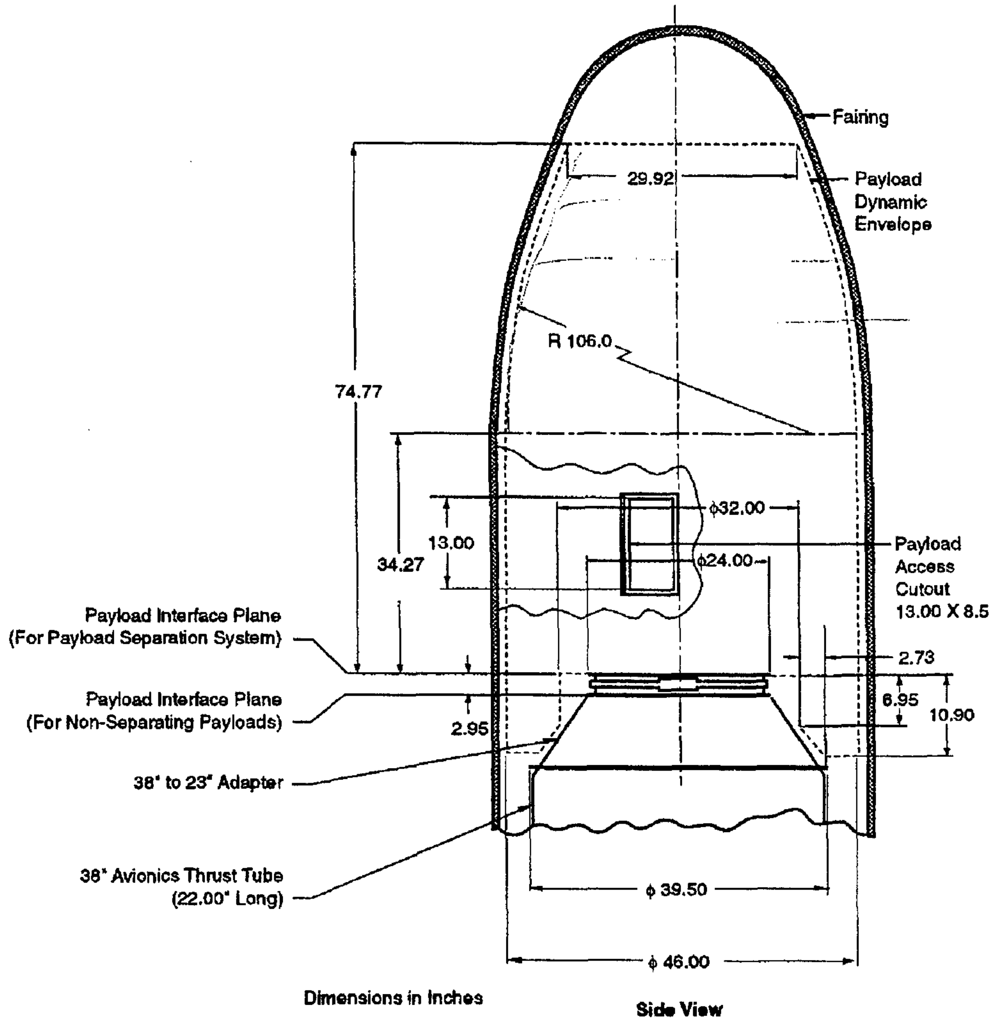
15.3.3 Payload Fairing

The fairing of the launch vehicle places size and shape constraints on the potential satellite design. Therefore, it is important to a satellite designer to understand these constraints. Pegasus XL offers two payload interfaces: a 38 inch diameter interface plate and a 23 inch diameter interface plate. Both of these interface plates can be used with or without HAPS. Figure 15-10 shows the payload fairing with the 38 inch interface. If this configuration is used with the optional HAPS, the spacecraft design will lose 14.65 inches from the available 84.22 inch height. Figure 15-11 depicts the 23 inch configuration without HAPS. When the HAPS is used with the 23 inch interface, 4.2 inches are deducted from the 70.57 inch height.



Source: OSC, 1993:5-3

Figure 15-10: Payload Fairing with 38 inch Interface



Source: OSC, 1993:5-4

Figure 15-11: Payload Fairing with 23 inch Interface

15.3.4 Hydrazine Auxiliary Propulsion System

The team questioned whether or not a HAPS should be included as part of the launch vehicle. The inclusion of a HAPS mandates additional restrictions on the height, mass and volume of the spacecraft, in a trade for added accuracy and/or higher orbit insertion. It was decided that the vast majority, if not all, of the Modsat missions would not require orbital altitudes above 600 kilometers. Additionally, orbital insertion accuracy is mission specific and depends on mass, targeted orbit, and the particular guidance strategy adopted for the mission (OSC, 1993:3-5). Thus, the team decided to exclude the HAPS configuration from consideration. This decision allowed greater flexibility in generating alternative designs for the satellite bus.

15.3.5 Other Considerations

Many other characteristics of the Pegasus XL influenced and constrained the design effort. These characteristics are listed in the Commercial Pegasus User's Guide. One of the constraints pertains to the spacecraft center of mass, which must be within 1.5 inches of the launch vehicle's center of mass in both the Y and Z axes. (Jim, we need a picture of the coordinate frame of reference here, if possible). The X-axis position of the spacecraft center of mass can not exceed 30 inches above the LV-to-spacecraft interface plate without severe restrictions being placed on the total mass of the spacecraft. (OSC, 1993:5-13)

Another factor to consider is that the spacecraft's stiffness in the Z-axis must be greater than 20 Hertz to avoid dynamic coupling with the launch vehicle (OSC, 1993:5-15). The maximum spacecraft mass is also limited due to critical shear stresses experienced at the interface plate. The maximum mass that can be integrated with the 38 inch interface plate is 454.6 kilograms (1000 lbm), while the maximum mass for the 23 inch interface plate is 318.2 kilograms (700 lbm) (OSC, 1993:5-9). Additional considerations are the axial and lateral loads experienced during the captive carry, drop, and launch environments. According to the Pegasus User's Guide, maximum axial accelerations can range from -4.2 to 13 times the force of gravity (-4.2 to 13G's). The maximum acceleration due to horizontal lateral loading is between +/- 1.5G's. The maximum vertical acceleration is between +/- 4.0G's.

To ensure that the design alternatives do not violate the Pegasus XL constraints, launch vehicle compatibility testing was incorporated into the Modsat model through 3-D surface rendering visualization (see Volume III).

15.4 Baseline Orbit and Allowable Launch Mass

15.4.1 Baseline Orbit-Type

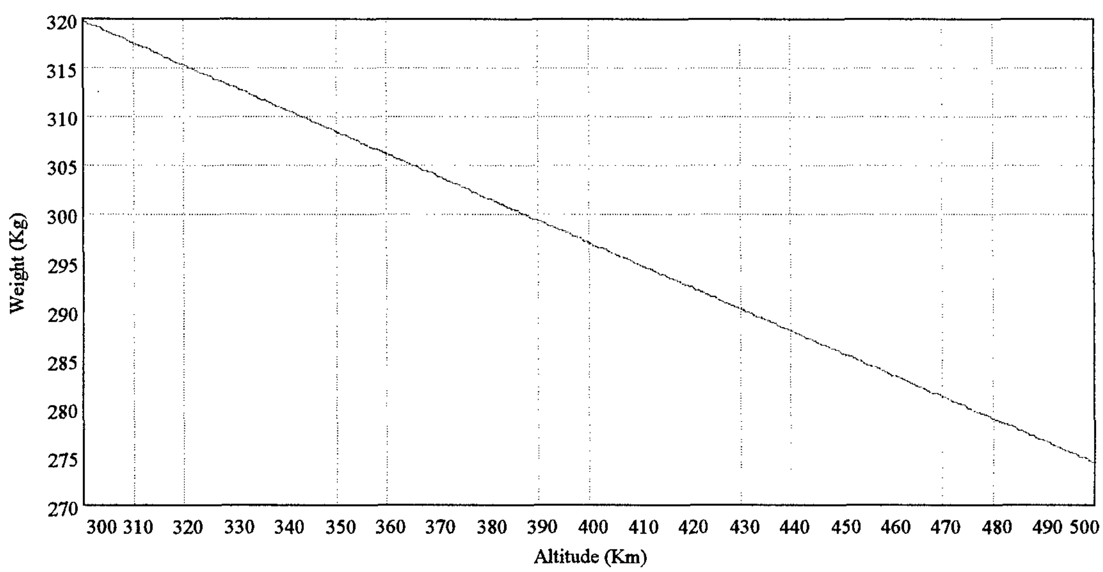
Section 15.3 demonstrated that sun-synchronous orbits are the most weight restrictive orbits for a launch vehicle, and many mission applications may require a sun-synchronous orbit. Therefore, the bus should be light enough to allow for such missions. But in order to establish a baseline launch mass, a baseline orbital altitude must be determined. This altitude will not be dictated to mission planners; in reality, they have the latitude to select an appropriate orbit, provided their spacecraft meet the launch mass constraint.

Nevertheless, the team needed a maximum mass limit for design purposes. Recall from the Value System Design section that one of the measures of effectiveness for this study is allowable mission module mass. For the baseline orbital altitude, this mass is the allowable launch mass less the mass of the bus. The challenge was to find the optimum sun-synchronous orbit for design, and therefore, the spacecraft mass limit. The optimum orbit is defined as that which maximizes the allowable launch mass.

15.4.2 Mass to Altitude Tradeoff

This section details the analysis that was performed in order to determine the optimum sun-synchronous orbital altitude for design purposes. It is essentially a study of the tradeoff between the mass carriage capacity of the launch vehicle for a given altitude and the amount of propellant to required to maintain that altitude for one year.

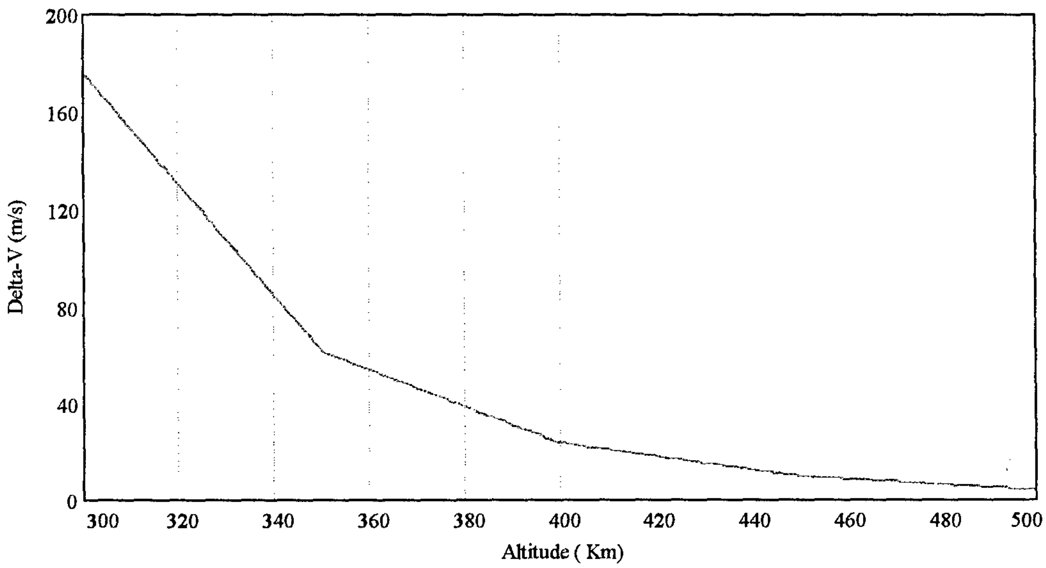
The mass-to-orbit performance of a launch vehicle decreases as the orbital altitude increases. Thus, lower altitudes are desirable for large payloads. The performance capability of the Pegasus LV for sun-synchronous orbits is shown in Figure 15-12.



Source: OSC, 1993:3-3

Figure 15-12: Launchable Mass Capacity for Sun-synchronous Orbits

However, as the orbital altitude decreases, the requirement for ΔV performance to maintain altitude increases. The reason for this is simple. At the lower altitudes, atmospheric drag on the spacecraft is significant. The spacecraft must provide thrust (ΔV) to counter this drag and maintain altitude. The ΔV vs. altitude curve is shown in Figure 15-13.



Source[1]: Larson and Wertz, 1992:back cover

Figure 15-13: The ΔV to Maintain Altitude per Year

Since ΔV is provided through the use of propellant, the amount of propellant on-board the spacecraft is proportional the ΔV required to maintain altitude. This statement assumes that the amount of propellant reserved for other uses (see section 15.5.2.1) does not depend on the altitude. In fact, for this study, it is assumed that the mass of the bus (constructed), with the exception of the propellant, is independent of orbital altitude. Thus, this analysis seeks to maximize the available launch mass, defined as the allowable

[1] ΔV to maintain altitude assuming a ballistic coefficient of $B_0 = m/C_dA = 100\ kg/m^2$.

launch mass for a sun-synchronous orbit, less the mass of the propellant reserved for altitude maintenance:

Available Mass = Launchable Mass - Propellant Mass

where the launchable mass for a given altitude is determined from Figure 15-12.

For the a given amount of ΔV, the corresponding propellant mass is given by (Sackheim and others, 1995:641):

$$m_p = m_f\left[e^{\left(\Delta V/I_{sp}g\right)} - 1\right] = m_o\left[1 - e^{-\left(\Delta V/I_{sp}g\right)}\right] \qquad \textbf{(Eqn 15-1)}$$

where $m_f = m_o - m_p$: the final vehicle mass
m_o : initial vehicle mass (launchable mass, found from Figure 15-12)
m_p : mass of propellant consumed
I_{sp} : specific impulse
g : gravitational constant which is 9.802 m/s^2

For this analysis, the chosen propellant is the hydrazine blend with the highest specific impulse (24% HN, 76% N_2H_4; I_{sp}=263 m/s) from among the mono-propellant database used in this study (see section 15.5.2.3.1).

By substituting this expression for propellant mass into the previous equation, the available launch mass becomes a function of allowable launch mass and ΔV, both of which are functions of orbital altitude. Thus, available launch mass is essentially a function of one variable, orbital altitude. It is desirable to find the altitude that maximizes this function. Figure 15-14 shows the function of available launch mass vs. orbital altitude. It is clear that this function is maximized at 350 km altitude, since the available mass increases with altitude until 350 km. Beyond this, the ΔV requirement decreases but the allowable launch mass decreases dramatically.

Thus, the baseline design altitude was chosen to be 350 km. Referring again to Figure 15-12, it can be seen that at this altitude the Pegasus can launch approximately 308 kg into sun-synchronous orbit. In satisfaction of the objective, "Maximize allowable mission module weight," this allowable mass is 308 kg less the mass of the bus.

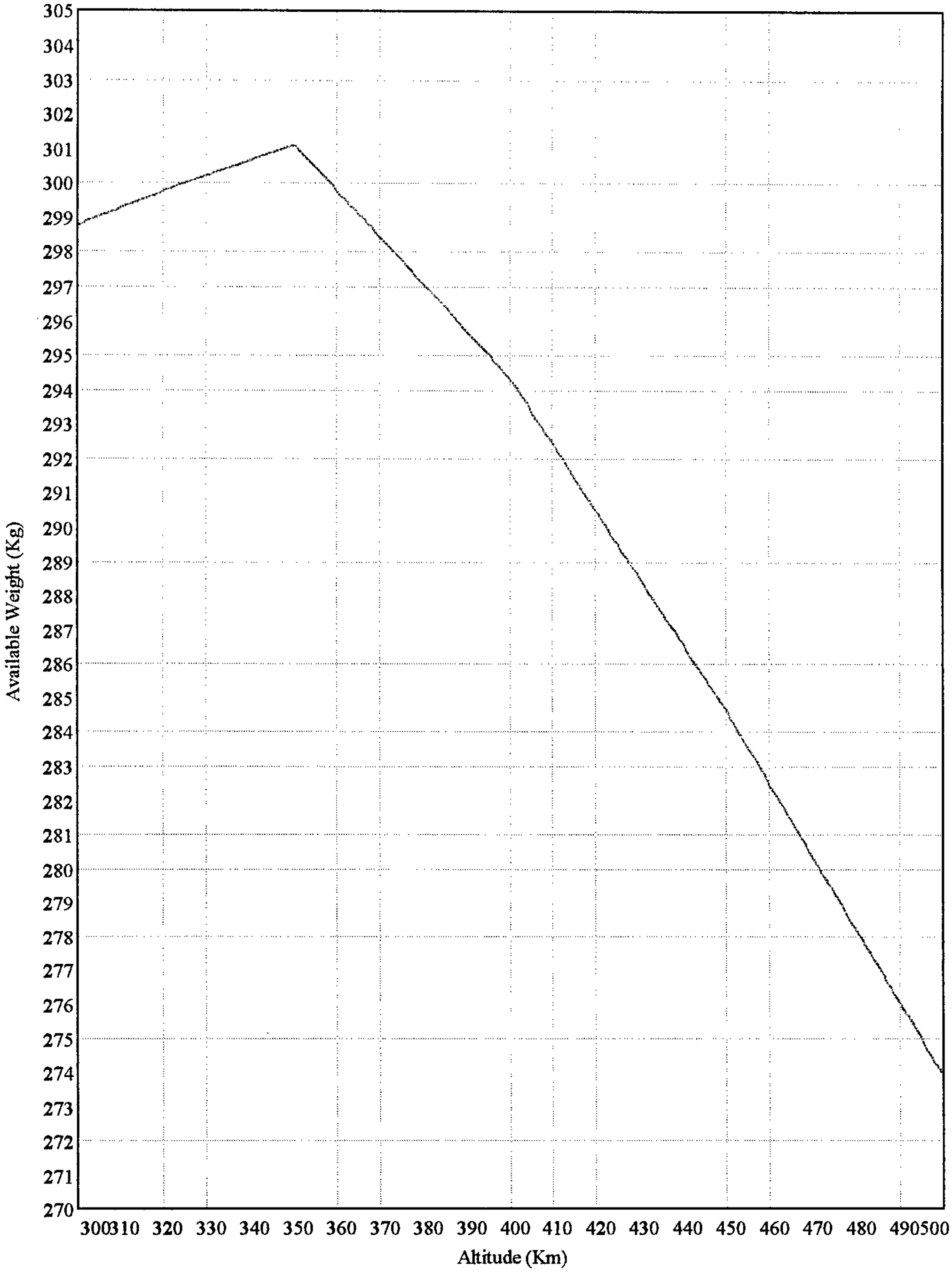


Figure 15-14: Available Mass for the Spacecraft

(Not Including Propulsion Subsystem)

15.5 Subsystem Level

Tradeoff analyses were performed at the subsystem level, in order to further narrow the solution space. The subsystems of Modsat are described below:

Table 15-9: Spacecraft Subsystems

Subsystem	Principal Functions	Other Names
Attitude Determination and Control System (ADCS)	Provides determination and control of attitude and orbit position, plus pointing of spacecraft and appendages	Attitude Control System (ACS), Guidance, Navigation and Control (GN&C) System, Control System
Propulsion	Provides thrust to adjust orbit and attitude, and to manage angular momentum	Reaction Control System (RCS)
Structures and Mechanisms	Provides support structure, booster adapter, and moving parts	Structure Subsystem
Thermal Control	Maintains equipment within allowed temperature ranges	Environmental Control System
Telemetry, Tracking and Command (TT&C)	Communicates with ground and other spacecraft; spacecraft tracking	Communication (Comm)
Electrical Power System (EPS)	Generates, stores, regulates, and distributes electric power	Power

Source: Reeves, 1992:287

15.5.1 Attitude Determination and Control

15.5.1.1 Introduction

The purpose of the Attitude Determination and Control Subsystem (ADCS) is to (1) stabilize the spacecraft against disturbing torques imposed by both the natural space environment and the spacecraft's own internal mechanisms, and (2) maneuver and orient the spacecraft in response to commands issued from the control station. Stabilization and control can be accomplished via multiple techniques. These include gravity-gradient, magnetic, pure-spin, dual-spin, one-axis bias momentum, and three-axis stabilization (Eterno, 1992:346). The ADCS itself can be grouped into three distinct sections: (a) attitude sensors, (b) actuators, and (c) control logic/control computers. Attitude sensors come in several varieties, including Sun sensors, Earth-horizon sensors, star sensors, magnetometers, and inertial measurement units (IMUs) (Eterno, 1992:360). There are also several types of actuators, including reaction wheels, momentum wheels, control-moment gyros (CMGs), electromagnetic torquers, and thrusters (Eterno, 1992:355). Each stabilization, sensing, and control technique has it's own advantages and disadvantages. The optimum combination of stabilization and control technique and ADCS depends largely on the spacecraft system performance requirements imposed on the ADCS, and to a lesser extent, constraints imposed by other satellite subsystems (including the payload). The sensors, control logic, and actuators are tied together with the satellite's dynamics and disturbance torques in the following basic manner:

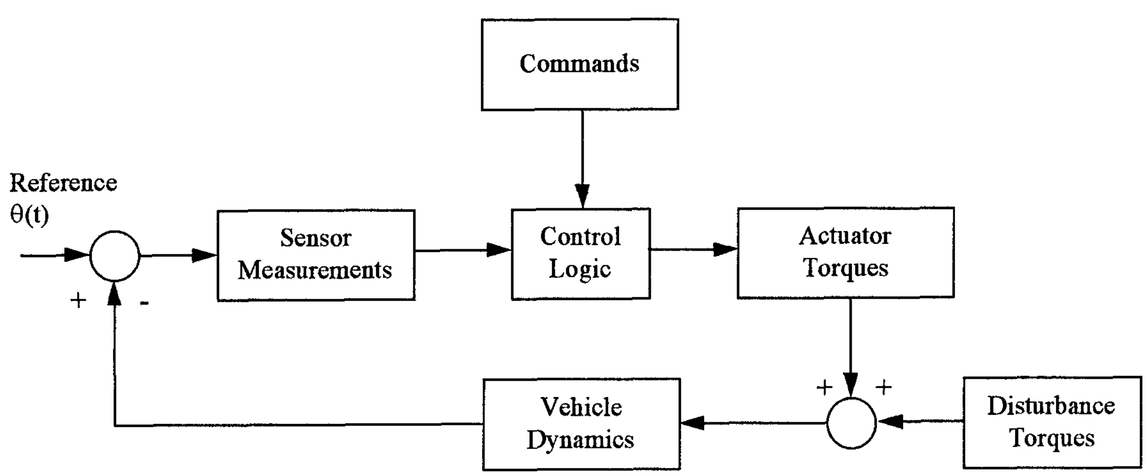


Figure 15-15: Simplified Attitude Control System

15.5.1.2 Requirements Definition

As put forth in the Task Statement provided by the chief decision maker (CDM), the spacecraft should be able to support four types of sensor payloads: electro-optical, infrared, laser designators, and synthetic aperture radar, and it should be able to support one-meter resolution imagery during an overhead pass. These mission support requirements typically drive the following spacecraft system performance requirements tied to ADCS performance requirements (Eterno, 1992:343):

- Pointing reference: Earth-pointing vs. Sun-pointing vs. Inertial-pointing
- Pointing range: Angular range over which the spacecraft must be able to orient itself with respect to the reference
- Pointing accuracy: Accuracy in which the spacecraft can be controlled to point in a specified direction
- Pointing knowledge: Accurately that the attitude sensors determine pointing angles and angular rates
- Pointing stability:
    - Jitter: Maximum allowable short-term, high-frequency motion the payload can tolerate and still perform it's mission
    - Drift: Maximum allowable limit tolerable on slow, low-frequency angular motion
    - Settling time: Time allowable for the spacecraft to recover and stabilize from maneuvers and other pointing disturbances
- Slew rate: Speed at which the spacecraft must be able to repoint from one direction to another

Sensor selection is driven primarily by pointing reference, knowledge, and accuracy requirements. Actuator selection will be driven primarily by pointing range, stability, and slew rate requirements. As previously mentioned, care must be taken so that constraints affecting other subsystems (for example, structural limitations) are not violated.

All requirements are ultimately traceable to the objectives set down in the Value System Design (VSD). The ADCS is affected by the fundamental VSD objectives "Maximize Tactical Responsiveness" and "Maximize the Mission Utility". Specifically, under "Maximize Tactical Responsiveness" attitude determination and control requirements are driven by "Maximize Capability for Tactical Maneuvers", as measured by slew rate, while "Maximize Pointing Accuracy" (a sub-value of "Maximize the Mission Utility") drives all other pointing requirements (knowledge, stability, etc.).

15.5.1.3 Design and Modeling Process

The fundamental ADCS design process used was the process developed by Eterno, Zermuehlen, and Zimbelman for the "Space Mission Analysis and Design" text (SMAD). As this process is useful for <u>estimating</u> spacecraft design specifications, it would need to be refined as the design evolves in order to incorporate more detail and greater accuracy. The actual mathematical modeling was done in a MATLAB environment using the equations provided by SMAD.

15.5.1.3.1 Control Modes

The first step in designing an ADCS was to define the different control modes during the mission. These were (1) Orbit Insertion, (2) Initial Attitude Acquisition, (3) Normal, (4) Slew, (5) Contingency/Safe, and (6) Special. Orbit Insertion is the period in which the spacecraft is brought into it's operational orbit, including the boost phase. Initial Attitude Acquisition begins after Orbit Insertion. During this phase, the spacecraft takes it's initial attitude measurements and uses them to stabilize itself. The Normal mode is the spacecraft's nominal, operational attitude mode, and is used for the vast majority of the mission duration. Slewing mode is simply the control mode in which the spacecraft performs repointing maneuvers. Contingency/Safe mode is used in emergencies, such as if Normal control mode is disabled, and usually involves the satellite orienting itself so that (a) the payload is protected, (b) the solar cells (if any) are positioned so that they receive the most sunlight, and (c) shutting down all non-critical functions and awaiting instructions from the ground station. The Special control mode can be defined as any mode not encompassed by the others. In this case, it will refer to when the satellite is configured for orbital altitude and inclination changes (Eterno, 1992:343). The requirements of each mode must be taken into consideration when selecting ADCS hardware.

During Orbit Insertion, tradable options were (a) no active spacecraft control, (2) simple spin stabilization, and (3) full spacecraft control using thrusters. In order to keep operations as simple as possible in keeping with a "tactical" mission, it was decided that the spacecraft will use no active control during orbit insertion but will rely on the booster for accurate and direct insertion into low-Earth orbit. Slight orbital insertion errors made by the booster can be corrected with Modsat's onboard Propulsion system. Initial Attitude Acquisition can be determined by the satellite's on-board sensor suite. Because this is intended to be a low-Earth orbiting spacecraft, an Earth sensor seems ideal for this control mode. The Earth sensor was selected because it can provide reasonable accuracy (< 1 degree) (Eterno, 1992:360) and is not vulnerable to eclipses as is a Sun sensor. Star sensors were not appropriate because they take time to accurately lock onto their targets. Often, at booster separation, some kind of rotational motion (i.e. "tip-off") is imparted to the spacecraft. In such a case, an IMU is useful because it can directly and accurately measure a satellite's rotational and translational rates. Control torque during Initial Attitude Acquisition can best be provided by thrusters. Although not useful for fine control, they can provide more than enough torque quickly enough to stabilize the spacecraft, in addition, fine control is not needed at this point. The hardware required for Normal mode can be any combination of sensors and actuators that meets the satellite's general pointing requirements. As long as slewing rate requirements are met, Slew mode can also use any combination of hardware. Contingency/Safe mode will use the same hardware as the Initial Attitude Acquisition control mode. Finally, during the Special mode, IMUs and thrusters should be able to provide reliable, accurate control during altitude and inclination changes.

15.5.1.3.2 Control Method

The next step in the ADCS design process is selecting the particular type of attitude control methods. Options include gravity-gradient, magnetic, pure-spin, dual-spin, one-axis bias momentum, and three-axis stabilization (Eterno, 1992:346). Since this decision essentially determined what kind of spacecraft was to be designed, it was decided at the system level to use a three-axis stabilized scheme. The primary reason behind this decision is gravity-gradient and magnetic control modes, while the simplest mechanically, do not provide the kind of fine pointing accuracy and control required by the proposed payloads (gravity-gradient ~ 5 deg, magnetic ~5 deg) (Eterno, 1992:346). Furthermore, both methods provide control on only two axes, unacceptable for the type of payloads envisioned for this satellite bus. Finally, gravity-gradient and magnetic control modes are subject to irregularities due to lack of detailed knowledge of the Earth's gravitational and magnetic fields, respectively. A spinning satellite was ruled out because it only provides control on two axes.

15.5.1.3.3 Disturbance Environment

The next step in the SMAD design process is to quantify the disturbance environment due to natural occurrences. There are four significant sources of natural disturbance torques: (1) the Earth's gravity (gravity-gradient disturbances), (2) solar radiation, (3) Earth's magnetic field, and (4) aerodynamic forces. (Eterno, 1992:353) Each is dominant at different orbits and under varying circumstances. Gravity-gradient effects are present at all orbits, but decrease with increasing orbital altitude. Its effects were estimated by the following relationship (Eterno, 1992:353):

$$T_g = \frac{3\mu}{2R^3}\left|I_{max} - I_{min}\right|\sin(2\theta) \qquad \textbf{(Eqn 15-2)}$$

T_g = max gravity torque (N*m)
μ = Earth's gravity constant (3.986 x 1014 m^3/sec^2)
R = orbit perigee (meters)
θ = max deviation of spacecraft z-axis from nadir vector (radians); assumed $\theta = \pi/4$ for max torque calculations
I_{min}, I_{max} = maximum and minimum moments of inertia (kg/m^2)
For the Earth-oriented application this design study focuses on, gravity-gradient torque is generally constant over an orbit. For a completely symmetric spacecraft, gravity-gradient torques are nonexistent, but can be significant for spacecraft with long, deployed

structures such as booms. The disturbance torque due to pressure from solar radiation is estimated by the following (Eterno, 1992:353):

$$T_{sp} = \frac{F_S}{c} A_S (1+q) \cos i (c_{ps} - c_g) \qquad \textbf{(Eqn 15-3)}$$

T_{sp} = max. solar radiation pressure torque (N*m)
F_s = solar constant (1358 watts/m^2)
c = speed of light (2.997 x 10^8 m/sec)
A_s = exposed spacecraft surface area (m^2)
q = overall spacecraft reflectance (dimensionless, estimated at 0.6)
i = angle of incidence to the Sun (assume i = 0 radians for max effect)
c_{ps} - c_g = max. distance between center of solar pressure and center of gravity (since c_{ps} and c_g were not well characterized, assumed difference is one-half the length of satellite body)

Torques generated by the effects of solar pressure are, unlike the other disturbances, generally independent of orbital altitude. This type of disturbance torque is highly dependent on the nature of the surface being illuminated, particularly on the radiation absorbtivity of the surface (absorbent, spectrally reflective, diffusely reflective, transparent, or some combination). The spacecraft's orientation with respect to the local sun angle also plays a significant role, as does the total spacecraft area exposed to the Sun, especially if extended solar panels are used. Solar torque is cyclic in nature, varying over the course of an orbit and nonexistent during an eclipse period. Torques caused by the Earth's magnetic field are dependent on the spacecraft's orbit altitude and inclination. The estimating relationship is (Eterno, 1992:353):

$$T_m = \frac{2DM}{R^3} \qquad \textbf{(Eqn 15-4)}$$

T_m = max. magnetic torque (N*m)

D = spacecraft's residual dipole (assume 1 ampere-turn*m^2)
M = Earth's magnetic moment (7.96×10^{15} tesla*m^3)
R = orbit perigee (meters)

The Earth's magnetic field is stronger at low-Earth orbits, almost nonexistent at higher orbits, and varies greatly with orbit inclination.

The final external source of disturbance torque is due to aerodynamics. It is estimated by (Eterno, 1992:353):

$$T_a = 0.5\left[\rho C_d A V^2\right]\left(c_{pa} - c_g\right) \qquad \textbf{(Eqn 15-5)}$$

T_a = aerodynamic torque (N*m)
ρ = atmospheric density at perigee (kg/m^3)
C_d = spacecraft drag coefficient
A = spacecraft cross-sectional area in direction of flight (m^2)
V = spacecraft velocity
c_{pa} - c_g = difference between spacecraft center of aerodynamic pressure and center of gravity (m) (since c_{pa} and c_g were not well characterized, assumed difference is one-half the length of satellite body)

Aerodynamic torques are nonexistent at high orbits, but can completely dominate all other disturbance torques at low orbits as indicated in Table 15-10. Aerodynamic torques, to the greatest extent, depend on the shape and size of the satellite and on the atmospheric density at a given orbit altitude. They are constant over the course of an orbit for Earth-oriented vehicles. It should be mentioned that, with the exception of gravity-gradient, all of the aforementioned forces vary with the solar cycle. Selection and sizing of the actuators primarily depends on the worst-case disturbance torques the spacecraft is expected to encounter. Other sources of disturbance torques are internal to the spacecraft (center-of-gravity uncertainties, thruster misalignments, fuel sloshing, etc.) and can be minimized by careful spacecraft design and tight manufacturing tolerances.

Table 15-10: Disturbance Torques

Source	Lowest (N*m)	Highest (N*m)
Gravity-gradient	1.6×10^{-6}	same
Solar pressure	4.0×10^{-6}	3.1×10^{-5}
Earth magnetic field	5.2×10^{-5}	same
Aerodynamics	0.0017	0.013

These calculations assume a 350km, Sun-synchronous orbit. The highest/lowest values are based on the orientation of the solar panels to the direction of flight. If they are facing the direction of flight, the aerodynamic loads on the satellite will be most severe. If they are orthogonal to the direction of flight, then aerodynamic loads will be the lowest. Similarly, the torque due to solar pressure will be more severe depending on the solar panels' incident angle to the Sun.

The analysis of Modsat's disturbance environment revealed that the major source of disturbance torques will be due to aerodynamics. Hence, for disturbance rejection, the ADCS will be designed to counteract aerodynamic torques.

15.5.1.3.4 Select ADCS Hardware

15.5.1.3.4.1 Sensor Selection

The next step, selecting and sizing the ADCS hardware, is the heart of the ADCS design process. The hardware must be selected in conformance with system performance requirements, it must be able to withstand and correct for disturbances, and at the same time it must not exceed it's size, weight, and power allocation. When considering sensors, the available options are Sun sensors, Earth-horizon sensors, star sensors, magnetometers, and IMUs. Since the spacecraft will not employ magnetic stabilization techniques, magnetometers were unnecessary. It was decided during an earlier design step that Earth sensors and IMUs would be a good idea. However, IMUs are subject to "gyro-drift", meaning they tend to lose accuracy over a long period of time, and must be periodically updated and corrected using information from other sensors. Earth sensors can be fairly accurate (< 1degree) (Eterno, 1992:360) but may not be accurate enough for laser designators and other extremely high-accuracy payloads. There are Sun sensors available that can achieve arc-second accuracy pointing knowledge, but are useless during eclipse periods, which occur often for low-Earth orbiting spacecraft. Star sensors can provide arc-second attitude knowledge (Eterno, 1992:360). They are fairly simple mechanically and computationally and are widely available. In addition, by including the Sun's stellar magnitude in it's database, they can theoretically double as Sun sensors. Therefore, Modsat will use a star sensor to meet its fine attitude knowledge requirements. Restating the choices made for Modsat's ADCS sensor suite, an Earth sensor, star sensor, and an IMU were selected. The size, weight, power, and performance of the selected sensors are listed in Table 15-11.

Table 15-11: Characteristics of Modsat Sensor Suite

Sensor	Model	Performance	Weight	Power	Supplier
IMU	YG9666B Miniature IMU	gyro drift rate = 0.02deg/hr	3.45 kg	33 watts	Honeywell
Earth sensor	13-105 LightSat Earth Sensor	accuracy = 0.04 degrees	2.41 kg	4 watts	EDO (Barnes)
Star sensor	CT-633 Stellar Attitude sensor	accuracy = 10 arcsec	2.49 kg	10 watts	Ball Aerospace

15.5.1.3.4.2 Actuator Selection

When selecting an actuator suite, the choices include reaction wheels, momentum wheels, control-moment gyros (CMGs), electromagnetic torquers, and thrusters (Eterno, 1992:355). The choice of actuators was determined by the amount of torque needed to meet attitude stability (itself driven by disturbance torques) and slewing requirements. Electromagnetic torquers, while simple and reliable, cannot generate enough torque for rapid slewing and momentum-dumping operations that a tactically-oriented satellite should have, are subject to uncertainties in the Earth's magnetic field, and may not work for some orbits and inclinations, particularly over the Earth's magnetic poles. Therefore, they will not be used.

It was previously determined that thrusters would be useful in the Initial Attitude Acquisition and Special control modes. Thrusters can provide plenty of force for both rapid slewing and momentum-dumping. They are also useful for coarse attitude control. This leaves either reaction wheels, momentum wheels, and/or CMGs for use in fine attitude control and resisting disturbance torques. CMGs operate by moving gimbals attached to a wheel spinning at a constant rate. The wheel has an angular momentum vector along one axis, the gimbal is turned about a second axis orthogonal to the first, and the satellite moves around the third orthogonal axis. CMGs are capable of generating large amounts of torque and can provide rapid, precise maneuvering for spacecraft. However, they tend to be heavy, expensive, and require a lot of power. They also require a fairly complex control law to operate (Eterno, 1992:356). Reaction and momentum wheels are essentially high-inertia rotors that rotate about a fixed axis. Momentum wheels turn in one direction only and have a nominal spin rate greater than zero, thus providing some inherent resistance to disturbance torques based on how large the wheel's angular momentum vector is. The rotor speed can be adjusted to compensate for disturbing torques. Reaction wheels have a nominal rotor speed of zero and can turn in either direction in response to disturbance torques. By turning the wheel, the equal and opposite reaction of the turning motor on the spacecraft rotates the spacecraft in the opposite direction. Reaction and momentum wheels don't provide the sheer torque capacity of CMGs, but are widely available and come in a variety of sizes. They tend not to require as much power as a CMG, either (Eterno, 1992:355). For the purposes of resisting disturbance torques and for low-speed slewing, reaction and/or momentum wheels are adequate, therefore either reaction wheels or momentum wheels will be used.

The pointing accuracy of reaction wheels depends on the design of the control logic (how finely can the wheels' rotation be controlled?) and the wheels' manufacturing quality (for example, are the wheels subject to "sticking" when started up or operating at slow speeds?). A momentum wheel's accuracy depends upon it's ability to generate an angular momentum vector large enough to resist disturbances via the gyroscopic "stiffness" a large angular momentum vector inherently provides. The appropriate size can be estimated by considering the largest expected disturbance a wheel must be able to counteract, calculating the slew torque a wheel must generate in order to meet slewing requirements, and the amount of angular momentum a wheel must be able to store based on accumulated disturbance torques (Eterno, 1992:357).

The largest expected disturbance a wheel is expected to be able to counteract is equal to the largest torque imposed on the satellite from the environment. From Table 15-10, the largest expected torque is 0.01318 N*m.

The slew capacity of a reaction wheel/momentum wheel was estimated by using (Eterno, 1992:357):

$$T = 4\theta\frac{I}{t^2} \qquad \textbf{(Eqn 15-6)}$$

T_{slew} = slew torque (N*m)
θ = max angle satellite must be able to slew through (radians)
I = largest value of spacecraft's inertia matrix (kg*m^2)
t = allowable time in which satellite must complete the slew (seconds).

The maximum angle and time was selected to be 180 degrees and 36 seconds, thus creating an slew rate requirement of 5 degrees per second. Sizing reaction wheels for angular momentum storage was estimated by the following equation (Eterno, 1992:357):

$$H = T_D P \qquad \textbf{(Eqn 15-7)}$$

H = wheel angular momentum storage capacity per orbit (N*m*s)
T_D = disturbance torque (N*m)
P = orbit period (seconds)

This assumes that the wheels will be desaturated (i.e. "momentum-dumped") an average of once per orbit. If more frequent desaturation is allowed, the wheels can be made smaller. If less frequent desaturation is desired, the wheels must be made larger.

To determine the amount of angular momentum a momentum wheel must generate to resist disturbances and maintain the required pointing accuracy (Eterno, 1992:357):

$$H = T\frac{P}{4\theta_a} \qquad \textbf{(Eqn 15-8)}$$

H = angular momentum (N*m*s)
T = disturbance torque (N*m)
P = orbit period (seconds)
θ_a = pointing accuracy (radians). Assume pointing accuracy equal to best pointing knowledge provided by sensors (10 arc-seconds)

Table 15-12 shows the required slew torque and angular momentum storage for both reaction wheels and momentum wheels.

Table 15-12: Required Slewing Torque and Angular Momentum Storage Capacity

Actuator	Required Slewing Torque (N*m)	Storage Capacity (N*m*sec)
Reaction Wheels	0.0009 N*m	9.34 N*m*s to 72.38 N*m*s
Momentum Wheels	0.0009 N*m	44590 N*m*s to 345600 N*m*s

The actual storage requirement is highly dependent on the orientation of the solar panels over the course of an orbit. It is unlikely that the satellite would be flown with the solar panels facing the direction of flight for the entire orbit (to do so may cause unacceptable degradation in the solar arrays), so the actual amount of required angular momentum storage should be much less than the maximum. Note that the momentum wheel storage requirement is extremely large (due to the large disturbances caused by aerodynamics in a 350 km orbit), hence for low orbits, momentum wheels are not a good choice. Reaction wheels should be used instead. Four reaction wheels, arranged in a pyramid, will be used. This type of arrangement requires that at least two wheels be actuated in order to rotate on a single axis. A simpler method would have been to align one wheel on each of the satellite's principal axes (hence eliminating the need for a fourth wheel), but by using a four-wheel configuration the wheels can be made smaller, and the combined effect of actuating two wheels at a time can generate as much or more angular momentum as with can be generated with one larger wheel. Table 15-13 contains a description of the reaction wheels selected.

Table 15-13: Characteristics of Selected Reaction Wheels

Name	T-Reaction Wheel Type B
Manufacturer	Ithaco
Angular momentum storage (each wheel)	16.6 N*m*s
Maximum torque (each wheel)	0.04 N*m
Size	25.5 cm diameter, 9.0 cm height
Mass	5.1 kg
Power	6.5 W (nominal), 17.5 W(peak)

Reaction wheels must be periodically desaturated, that is, the stored angular momentum must be disposed of. To do this, thrusters will be used. Thrusters must be sized so that, in addition to being able to provide translational motion to the spacecraft, they can provide enough torque to hold the spacecraft steady while the wheels are desaturated, and to skew the spacecraft if necessary. The equation for sizing a thruster to meet slew requirements of a reaction wheel-driven spacecraft is (Eterno, 1992:358):

$$F = \frac{I}{Lt}(slew\ rate) \qquad \textbf{(Eqn 15-9)}$$

F = thrust force needed to meet slew requirement (N)
I = max. moment of inertia ($kg*m^2$)
L = thruster moment arm (m)
t = time required to accelerate from zero to 5 degrees/sec in 3 seconds

For Modsat, the required force level is approximately 0.01 N, well within the capability of most thrusters.

The estimated thruster size to dump momentum is given by (Eterno, 1992:359):

$$F = \frac{H}{Lt} \qquad \textbf{(Eqn 15-10)}$$

F = force level required (N)
H = stored angular momentum (N*m*s)

L = thruster moment arm (m)
t = thruster burn time (sec)

For Modsat, allowing two minutes to desaturate the reaction wheels, the required thruster size is 1.1 N. If more time is allotted for momentum dumping, thruster size can be decreased. If less time is required, thruster size must be increased. The thruster suite ultimately selected for Modsat included six 22.4 N thrusters (see section 15.5.2), which allows momentum to de dumped in less than six seconds per wheel.

15.5.1.3.5 Control Algorithms

The design of the control algorithms was beyond the scope of this project, and not relevant to any of the decisions made.

15.5.2 Propulsion

Space propulsion systems perform three major functions. They lift the launch vehicle and payload from the launch pad and place the payload into low-Earth orbit (LEO). They transfer loads from low-Earth orbits into higher orbit or into trajectories for planetary encounters. Finally, they provide thrust for attitude control and orbit corrections.

Selection of the launch vehicle has been previously discussed, and Modsat will not require propulsion to transfer orbits, due to its LEO requirement. Therefore, this section of the report will focus on the third major function listed above, that of an on-board spacecraft propulsion subsystem.

The main objective of this propulsion subsystem design study was to provide a cost-effective system, in accordance with the requirements and objectives of the overall study. Since a major emphasis has been placed on the use of competitively priced, proven technology, the team did not set out to design new concepts in spacecraft propulsion. Instead, the design effort was limited to making use of currently available, off-the-shelf technology.

15.5.2.1 Propulsion Functions

The Modsat propulsion subsystem performs the following functions:

- Orbital corrections.
- Tactical re-deployment.
- Acquisition of Sun, Earth, or star.
- On-orbit back-up mode control with 3-axis stabilization.
- Momentum management.
- 3-axis control during ΔV.

Note that during normal operations attitude control will be primarily performed by the Attitude Determination and Control Subsystem (ADCS). The propulsion system serves as a backup option in case the ADCS is off-line.

15.5.2.2 Propellant and ΔV Budget and Thrust Levels

Most of the necessary calculations are discussed in the ADCS subsystem study, section 15.5.1. Total propellant mass is calculated with respect to the chosen propellant and the demanded ΔV capability. Thruster size depends on the maximum required amount of acceleration.

15.5.2.3 Propulsion System Options

There are three candidate propulsion options for this study: cold gas, liquid mono-propellant, and liquid bi-propellant. The comparisons among them are shown in Table 15-14 and Table 15-15. Although cold gas has attractive features such as simplicity, high reliability and low cost, its limited specific impulse, low thrust ratio, and heavy weight led to its elimination from consideration. The remaining material focuses on the liquid mono-propellant and bi-propellant options.

Spacecraft propulsion systems have five major components:

- Propellant : The fuel of the propulsion system.
- Fuel feed systems: Feeds the propellant distribution system with fuel on demand.
- Storage Systems (tanks): Propellant storage.
- Engine: Thruster.
- Connections: System that connects the other components and manages fuel distribution through the use of pipes, vanes, regulators etc. The design of a specific connection system is beyond the scope of this preliminary design study. Rather, all candidate propulsion system solutions will account for connections as a percentage of cost, weight , volume etc.

Table 15-14: Candidate Propulsion Options

Type	Energy	Advantages	Disadvantages
Cold Gas	High pressure	1.Simple hardware. 2.High reliability. 3.Low cost. 4.Perfect when the exhaust temperature is important for satellite.	1.Very low performance(e.g. specific impulse and thrust). 2.The heaviest of all systems for given performance level.
Liquid Mono-propellant	Exothermic decomposition	1.Simpler than bi-propellant systems. 2.More reliable than bi-propellant systems. 3.Lower cost than bi-propellant systems.	1.Lower performance(e.g. specific impulse and thrust) than bi-propellant systems. 2.Higher weight than bi-propellant.
Liquid Bi-propellant	Chemical reaction	1.Higher performance (e.g. specific impulse and thrust). 2. MMH and UDMH[2] are storable.	1.Cyrogenic material handling difficulty. 2.Some propellants are dangerous. 3.Complicated systems

Source: Sackheim and others, 1992:645-646

Table 15-15: Other Propulsion Options

Type	**Description**
SOLID	One shot propulsion; usually used for orbit insertion in launch systems
LIQUID	
Water Electrolysis	Complicated; not developed for commercial use; high power requirement
Hybrid	Bulkier than solid systems; usually used for orbit insertion and orbit maintenance, not for attitude control
ELECTRIC	
Electrothermal Electrostatic Electromagnetic	High power requirement; complicated; some systems have high development risk

Source: Sackheim and others, 1992:645-646

[2] Monomethyl Hydrazine (MMO) and Unsymmetrical Dimethylhydrazine (UDMH)

15.5.2.3.1 Propellant

The most common liquid propellant is hydrazine, which is flight proven and experienced. All its properties, hazards, safety facts, storage conditions, etc. are well known. Six different blends of mono-propellants (Table 15-16) and five different hydrazine bi-propellant combinations (Table 15-17) were examined as alternative options and were taken into consideration in all calculations to generate alternative propulsion options.

A mono-propellant contains an oxidizing agent and combustible matter in a single substance. It may be a mixture of several compounds or it may be a homogeneous material, such as nitromethane or hydrazine. Mono-propellants are stable at ordinary atmospheric conditions but decompose and yield hot combustion gases when heated or catalyzed.

Table 15-16: Mono-propellant Blends

Freezing Point (K)	Composition WT. - %			Density (gm/cc)	Decomposition Temperature (K)	I_{sp} (sec)
	HN	H_2O	N_2H_4			
274.67	00	00	100	1.004	1,344	245.9
252.45	18	8	74	1.080	1,422	243.0
239.12	20	12	68	1.093	1,360	236.2
254.67	24	00	76	1.110	1,622	263.8
239.12	24	09	67	1.109	1,510	245.8
219.12	25	17	58	1.120	1,300	229.9

Source: Hydrazine Handbook, 1995:2-18

A bi-propellant rocket unit has two separate propellants, an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. The majority of liquid propellant rockets have been manufactured for bi-propellant applications.

Table 15-17: Bi-propellant Combinations

Combination		Mixture Ratio		Density	Chamber Temperature	I_{sp}
Oxidizer	Fuel	By Mass	By Volume	(gm/cc)	(K)	(sec)
Oxygen -	Hydrazine	0.90	0.80	1.07	3027	313
Oxygen -	UDMH	1.65	1.14	0.98	3321	310
Fluorine -	Hydrazine	2.30	1.54	1.31	4408	363
Nitrogen Tetroxide _	Hydrazine	1.34	0..75	1.22	2977	292
Nitrogen Tetroxide _	50% Hydrazine 50% UDMH	2.00	1.24	1.21	3088	288

Source: Dueber and McKnight, 1993:139

A cryogenic propellant is liquefied gas at low temperature, such as liquid oxygen (—147°C) or liquid hydrogen (—253°C). Provisions for venting the storage tank and minimizing vaporization losses are necessary with this type.

Storable propellants (e.g., nitric acid or gasoline) are liquid at ambient temperature and can be stored for long periods in sealed tanks. Space-storable propellants are liquid in the temperatures of space; this storability depends on the specific tank design, thermal conditions and tank pressure.

The propellant mixture ratio is the ratio of oxidizer to fuel in the reaction mixture.

15.5.2.3.2 Fuel Feed Systems

The following material is based largely on the work of Sackheim, Wolf and Zafran (1992: 654) and Sutton (1992: 211-230). The selection of a particular feed system and its components is governed primarily by the following: the application of the rocket; its size; type and amount of propellant; required thrust; flight program and duration; number or type of thrust chambers; past experience; mission velocity; and in general the need for simplicity of design, ease of manufacturing, reliability of operation, and minimum weight.

In general, a pressure feed system delivers superior performance over a turbo-pump system, when the total impulse or the mass of propellant is relatively low, the chamber pressure is low, the engine thrust-to-weight ratio is low (usually less then 0.6), and when there are repeated short-duration thrust pulses. The heavy-walled tanks for the propellant and the pressurizing gas usually constitute the major weight of the engine system. In turbo-pump feed systems the propellant tank pressures are much lower (by a factor of 10 to 40); thus, the tank weights are much lower (again by a factor of 10 to 40). Turbo-pump systems usually deliver superior vehicle performance when the total impulse is large (higher ΔV) and the chamber pressure is high.

The pressurized feed system can be relatively simple, such as for a single-operation, factory-preloaded, simple unit (with burst diaphragms replacing some of the valves). On the other hand, pressurized feed systems can also be quite complex, as with multiple re-startable thrusters or reusable systems.

System options are:

- Pressure-fed Systems: Blow-down or regulated systems
    - Used for rockets that deliver low (up to 10 N) or moderate (10 N to 200 N) levels of thrust and total impulse.
    - The overall system weight is reduced due to the simplicity of this system.

- They are reliable, due to their simplicity.

- Pump-fed Systems: Piston or turbo-pump, which can be driven by turbines, gas pressure intensifiers, or directly by electric motors.
  - For large total impulse and thrust requirements, the propellant tanks may become prohibitively large and heavy.
  - Heavier than pressure-fed systems.
  - Definitely lighter for applications using high thrust and longer total burn, such as launch vehicles.

The choice of fuel feed system type was influenced heavily by the fact that the bus design must support small tactical mission modules during a relatively short mission lifetime, and that the bus will be launched on the Pegasus Launch Vehicle. Given these considerations, a pressure feed system is a reasonable choice, for the following reasons:

- The propulsion system will be used for altitude maintenance, momentum dumping and inclination change (for tactical maneuvers). There is no known need for high thrust and high burn times.
- The bus system has been designed for one lifetime; reusability is not considered.
- The bus must support a small satellite. Weight and volume considerations were very influential during design. Thus, the small and light pressure-fed systems are appropriate.

One of the simplest and most common means of pressurizing the propellants is to force them out of their respective tanks by displacing them with high-pressure gas. This gas is fed into the propellant tanks at a controlled pressure, thereby giving a controlled propellant discharge. Because of their relative simplicity, rocket engines with pressurized feed systems tend to be very reliable. With mono-propellants, the gas pressure feed system becomes simpler than for bi-propellants, since the number of pipes, valves, and tanks is reduced.

Blow-down and regulated pressure systems were considered as viable options throughout the design effort; see Figure 15-16.

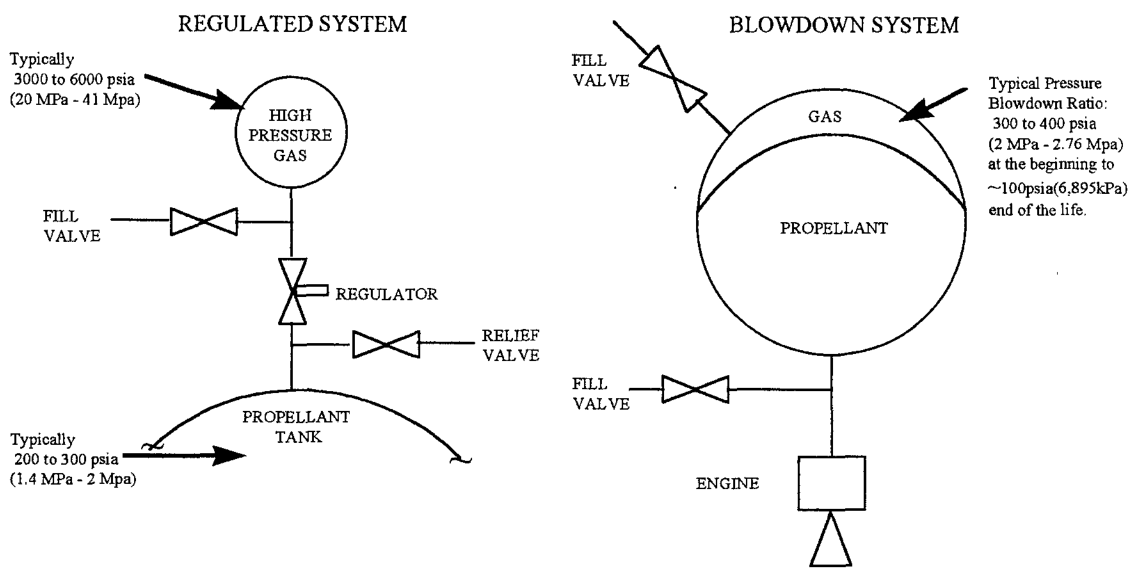


Figure 15-16: Gas Pressure Systems

Source: Sackheim and others, 1992:654

15.5.2.3.3 Storage Systems

Liquid storage systems need to manage the liquid propellant under zero gravity to ensure that neither gas nor vapor is expelled from the tank. Options for liquid propellant management include:

- Artificial gravity, induced by spinning spacecraft. This can be ruled out, since Modsat will be 3-axis stabilized.

- Inducing propellant burn by the use of a small rocket.
- Positive expulsion systems: use an active element (a bladder, piston, or bellows) to separate the pressurant gas from the liquid propellants under all dynamic conditions, and to force the liquid from the tank into the feed lines on demand.
- Surface tension systems: passively manage the propellants in a zero gravity environment by using vanes, screens, or sponges to wick the propellant into the propellant tank outlets. The pressurizing gas bubble is always kept in the center of the tank. All of these devices rely on surface tension forces to separate liquids from gases.
- Piston expulsion devices: permit the center of gravity (CG) to be accurately controlled and its location to be known. This is important in rockets with high side accelerations such as antiaircraft missiles or space defense missiles, where the thrust vector needs to go through the CG. If the CG is not well known, unpredictable turning moments may be imposed on the vehicle. A piston also prevents sloshing or vortexing.

The positive expulsion system is appropriate for Modsat, for the following reasons:

- The ADCS requirement for accurate control.
- They are flight proven, off-the-shelf designs.
- They have good expulsion efficiency.
- They are lighter than piston expulsion systems.

Among positive expulsion systems there are several alternatives, but at this level of design they will not be considered. It is assumed that the most common diaphragm will be used as a storage system. (Sackheim and others, 1992:655; Sutton, 1992:216-223).

15.5.2.3.4 Engine (Thruster)

Depending upon specific mission requirements, a spacecraft can employ from a few thrusters to more than 30 thrust units. These thrusters may generate long steady-state burns or short pulses of several milliseconds, each used to control maneuvers, maintain orbit, control attitude or manage momentum.

Two primary considerations in choosing thrusters are operating life (total number of pulses), thrust, total impulse, and time of steady state firing (sec).

It was not the intent of this study to design new thrusters. The approach was to examine the overall objectives for the Modsat bus and to select the appropriate engine type and number from the available off-the-shelf items.

15.5.2.4 Designing a Propulsion System

To design a propulsion system, one must analyze requirements, tradeoff design features, and size the system iteratively until the best configuration forms for a particular mission. The analysis of requirements should include performance, interfaces, and physical characteristics. Design tradeoffs should investigate different types of propulsion systems, selection criteria, and design factors for the specific mission. Through use of sizing calculations and a safety factor, types and quantities of the various propulsion components can be determined, including propellants, tank configuration, basic subsystem components, instrumentation, power conditioning, and pressurants. All decisions must be made within the context of the requirements and objectives.

The propulsion system elements are affected by several design factors, as shown in Figure 15-17.

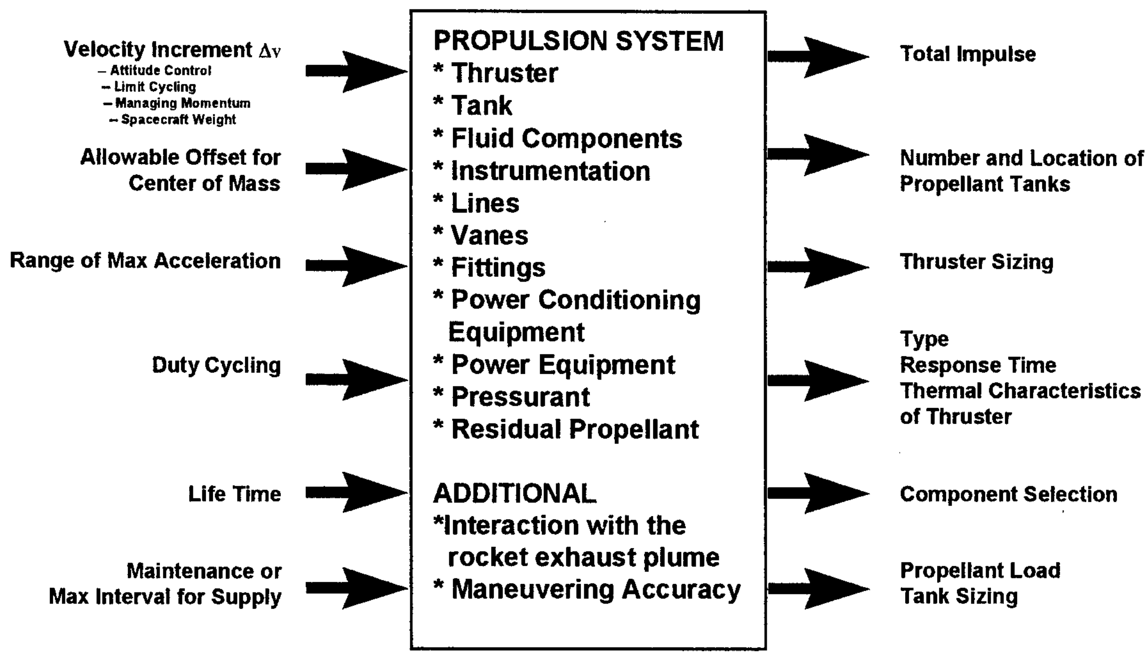


Figure 15-17: Propulsion Design Factors

Source: Sackheim and others, 1992:660-661

15.5.2.4.1 Design Basics

(1) *Tank Location*: Must reside at or near the spacecraft's center of mass to avoid shifting of the center of mass due to propellant usage.

(2) *Engine Location*:

(a) *Translational Control*: Engines for translational control are aligned to thrust through the center of mass.

(b) *Attitude Control*: Engines for attitude control thrust tangentially, and are mounted as far away from the center of mass as possible to increase the lever arm, thus increasing the torque per unit thrust.

(3) *Number of Engines:* Attitude control thrusters that fire in the direction of flight (along or opposite the direction of the velocity vector) are generally used in pairs to produce a pure torque without net linear force. Three-axis control requires a minimum of six attitude control thrusters; many designs use eight to twelve plus backup units for reliability.

15.5.2.4.2 Propellant Mass

For a typical design, the propellant load is sized based on information provided from the propellant budget table. For initial sizing, the factors to consider are velocity correction and control, and attitude control. Propellant load, m_p, can be estimated from the total impulse requirements (Sackheim and others, 1992:641):

$$m_p = m_f\left[e^{\left(\Delta V/I_{sp}g\right)} - 1\right] = m_o\left[1 - e^{-\left(\Delta V/I_{sp}g\right)}\right] \qquad \textbf{(Eqn 15-11)}$$

where $m_f = m_o - m_p$: the final vehicle mass
m_o : initial vehicle mass
m_p : mass of propellant consumed
R : mass ratio (m_o / m_f)
I_{sp} : specific impulse
g : gravitational constant.

Knowing the propellant mass, one can determine the propellant volume, V_p, by dividing by propellant density, d_p (i.e., $V_p = m_p / d_p$).

15.5.2.4.3 Propellant and Pressurant Volume for Tank Sizing

The tank may be sized after determining the propellant load. For bi-propellant systems, the oxidizer and fuel requirements are determined from

$$\mathbf{O / F = m_{ox} / m_{fuel}} \qquad \textbf{(Eqn 15-12)}$$

where O / F is the oxidizer to fuel mixture ratio needed to deliver the required specific impulse, m_{ox} is the oxidizer mass, and m_{fuel} is the mass of the fuel. Tank volumes are calculated from:

- Propellant volumes loaded into each tank
- Reasonable allowance for ullage (gas volume in the propellant tank, approximately 5%) (Sackheim and others, 1992:659)
- Design margin.
- Propellant remaining in each tank because of trapped liquids or uncertainties in loading and performance.

The mixture ratio is a critical parameter in propulsion system sizing; therefore its selection is the first step in determining the quantity of propellant needed to satisfy total impulse requirements. There are different O / F ratios for different bi-propellant couples. A good rule of thumb is to use equal size tanks, since this simplifies tank manufacturing, subsystem configuration layout, and integration into the spacecraft.

For a mono-propellant system, the above discussion is valid, except that there is no oxidizer.

Pressurant gas requirements depend on the type of pressurization system employed: either regulated, blow-down or a combination of the two (Figure 15-16: Gas Pressure Systems)

- Regulated: About 5-10% ullage is provided in the propellant tanks. The tank volume will be almost the same as the propellant volume, since the pressurant is stored in the different tank.
- Blow-down: The total propellant tank volume for the blow-down system is the sum of the propellant volume, V_p, and initial gas volume, V_{gi}, in the tank. The blow-down ratio is (Sackheim and others, 1992:659):

$$\mathbf{R = V_{gf} / V_{gi} = [\,V_{gi} + V_p\,] / V_{gi}} \qquad \textbf{(Eqn 15-13)}$$

where V_{gf} is the final gas volume, neglecting the propellant volume remaining at the end of the life as well as density changes with temperature.

One can allow for a liquid propellant load design margin of about 25%, and a reasonable residual propellant of 5% for early conceptual design.

For most systems, one can determine the pressurant mass from the perfect gas law, but only when the propellant is withdrawn isothermally (in blow-down systems at low duty cycles). Otherwise, calculating the requirements for pressurant mass for regulated systems can become thermodynamically complicated. By using conservation of energy, the pressurant mass can be determined (Sackheim and others, 1992:660):

$$m_{gi} = \frac{P_p V_p}{R T_i}\left[\frac{k}{1-\left(P_g / P_i\right)}\right] \qquad \textbf{(Eqn 15-14)}$$

where

m_{gi} : initial pressurant mass
P_p & V_p : instantaneous gas pressure and volume *in the propellant tank*
P_g (300-600 psia)&P_i(3000-6000 psia): instantaneous gas pressure and initial gas pressure *in the pressurant tank*
T_i : initial gas temperature (275-300 K)
k : specific heat ratio for pressurant gas (1.40 nitrogen, 1.67 helium etc.)

R : pressurant gas constant (296.8 J/(Kg.K) for nitrogen, 2077.3 J/(Kg.K) for helium etc.)

Note: This equation does not apply to very high storage pressures, for which the compressibility factor becomes important. In this case, the volume of the pressurant can be found by its density.

One may estimate the propellant and pressurant gas tank weights from (Sackheim and others, 1992:660):

$$\sigma = \frac{pr}{t} \text{ (cylindrical)} \qquad \textbf{(Eqn 15-15)}$$

and

$$\sigma = \frac{pr}{2t} \text{ (spherical)} \qquad \textbf{(Eqn 15-16)}$$

where σ : allowable stress
p : maximum expected operation pressure
r : tank radius
t : tank thickness

For a typical spacecraft propulsion system, the tank weight will be about 5 to 15% of the propellant weight, depending on the basic design, safety factors, and construction materials. Also, one must add 20 to 30 % of the overall tank weight for mounting and propellant management devices (Sackheim and others, 1992:660).

Liquid propulsion systems are typically 85-93% propellant by mass, with the remainder consisting of pressurant, thrusters, tanks, fluid components, lines, fittings and instrumentation (Sackheim and others, 1992:660). This fuel mass fraction, however, can be considerably lower in small systems, such as Modsat. Higher fuel mass fractions are usually associated with large propellant loads and the use of composite, over-wrapped hi-tech tanks.

15.5.2.5 Interfaces Among Subsystems and Other Considerations

The interfaces among the subsystems are shown in Figure 15-18. The following factors must be considered:

- Thruster exhaust plume
  - (a) Plume heating of surfaces next to the thruster.
  - (b) Forces and moments that the plume places on the spacecraft. For example, solar panel-related drag may occur. To avoid this, the thrusters must be mounted away from the solar panels.
  - (c) Contamination by plume, which is dependent on propellant type, may affect the sensitive surfaces (such as optics, solar cells, thermal control surfaces, I/R sensors, etc.) of the satellite. Contamination is a serious concern if the plume reduces the solar cell efficiency and overall available electric power.
- Maneuver accuracy.
- The propulsion subsystem does not use much electrical power, unless it employs thrusters with heated catalyst beds, known as heated thrusters. Also, propulsion lines and tanks must be protected from freezing, usually by thermostatically controlled guard heaters. Power for these heaters is included in the thermal subsystem.

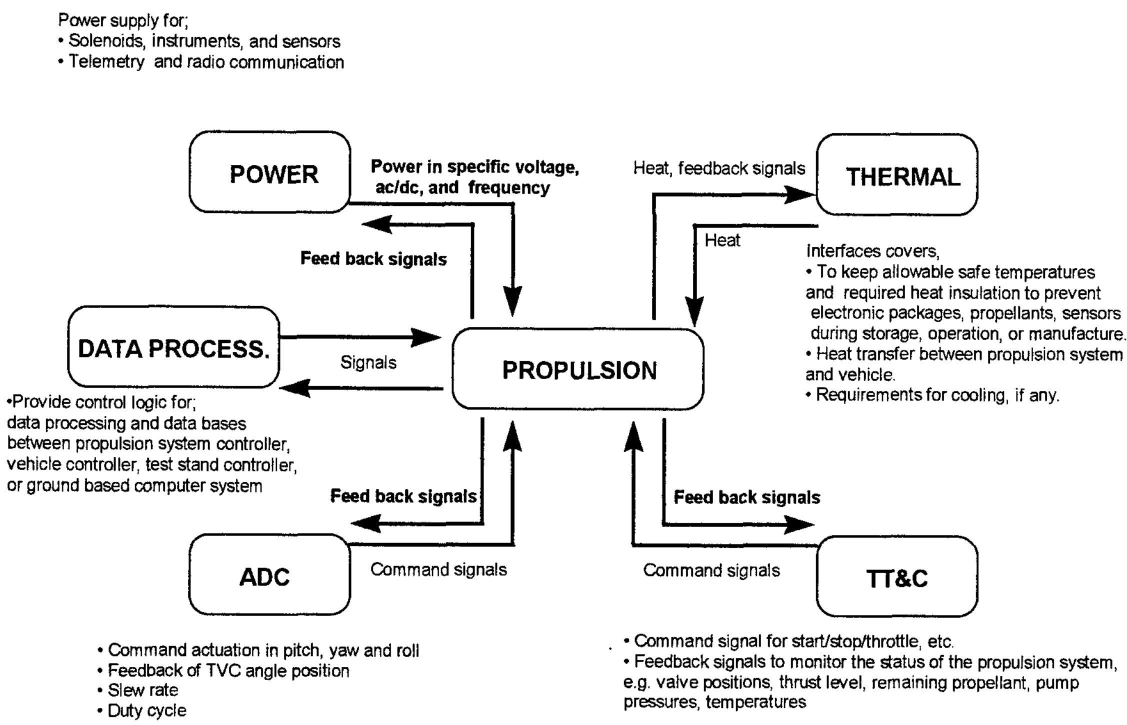


Figure 15-18: Propulsion Block Diagram

15.5.2.6 Propulsion Options For Standard Satellite Bus Alternatives

Propulsion subsystem options for Modsat were modeled and chosen with the aid of the Modsat computer model. The primary alterables for the propulsion subsystem are shown in Table 15-18.

Table 15-18: Propulsion Alterables

Alterable Elements	**Possible Options**	
Propulsion Type	Liquid mono-propellant	Liquid bi-propellant
Pressure System	Blow-down	Regulated
Propellant type	Many	
Shape of Tanks	Spherical	Cylindrical

The design and placement of the Modsat propulsion subsystem is constrained primarily by the volume and weight limitations of the Pegasus Launch Vehicle. Therefore, volume and weight considerations are major criteria for the selection of propulsion alterables.

15.5.2.6.1 Propulsion Subsystem Location

It was decided that the entire propulsion subsystem should reside on the bottom level of the satellite bus. The generic, modular nature of the bus precludes any other location for this subsystem. This major design decision was made after discussing the following considerations:

- The propellant tanks of the propulsion system are some of the heaviest components of the satellite bus. Launch vehicle considerations deem that the center of gravity of the spacecraft be as low as possible on the LV/spacecraft z-axis.
- Since the specific mission modules that will be flown on Modsat are unknown, it is wise to keep any thrusters well below the bus-to-mission module interface.
- For some of the candidate system solutions, there is free space for the insertion of additional components. Therefore, the final configuration of the bus is unknown. This makes it difficult to place valves and thrusters anywhere above the bottom of the bus. Even in the candidates systems that don't have the free space, it is quite difficult to locate propulsion equipment amid the other subsystems.
- The effects of thruster exhaust plume and other possible contamination from the propulsion system are reduced by locating it on a designated bottom plate.

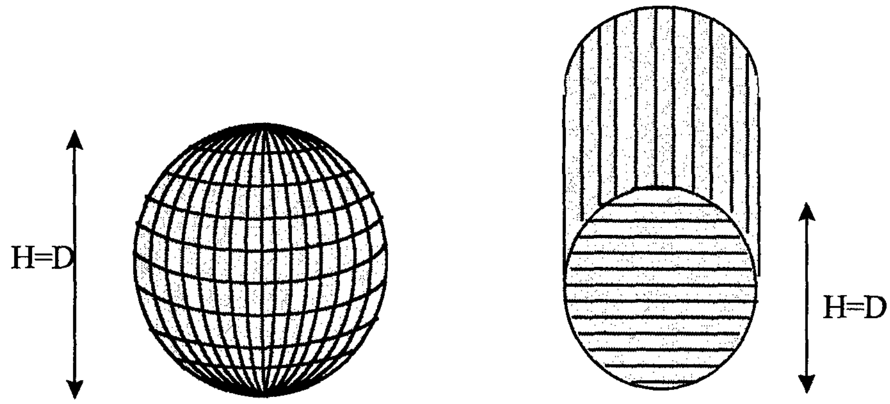
- The propulsion subsystem does not fit in well with the modular nature of the bus, with its removable components and structural assemblies (see section 15.5.3). Propulsion equipment includes relatively large structures (tanks) and complicated fuel delivery systems that do not lend themselves to modularity, or removal and replacement. It would be extremely difficult to locate any of these components and systems anywhere other than on a fixed plate at the bottom of the bus.
- The weight of pipes, vanes, valves, and the other propulsion plumbing equipment is reduced by consolidating the whole subsystem in one location.

Despite these considerations, the Modsat model provides the engineer with maximum flexibility in the placement of subsystems and components. Future efforts could examine other placement schemes for the propulsion subsystem.

15.5.2.6.2 Propellant Tank Geometry

The bottom level of Modsat is dedicated for the propulsion subsystem, with the exception of one SGLS antenna. This level is an octagon of approximately 86 cm diameter; a 5 cm square structural spine runs through the center (see section 15.5.3; see also Volume III). Note that this spine precludes a centered one-tank configuration. The location of the satellite center of mass is critical, both in the launch phase or during on-orbit operations. The tank(s) should be balanced with respect to the central axis of the spacecraft, and must be placed as near as possible to this axis.

For simplicity, and in order to make efficient use of the space in the bottom level, it was decided that the tanks should have a common shape. Although the most common tank shape is a sphere, cylindrical tanks are also acceptable and reasonable. The tanks should lie on the bottom, in order to minimize the total height occupied by the tanks (see Figure 15-19).



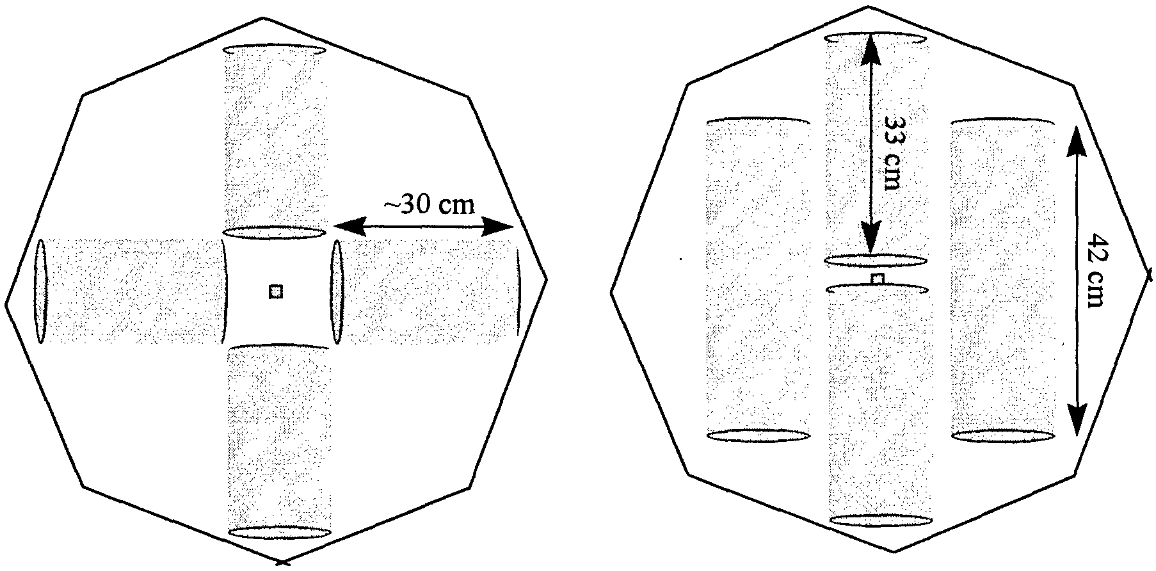
Height = Diameter of sphere   Height = Diameter of cylinder

Figure 15-19: The Difference in Height Between Spherical and Cylindrical Tanks

Initially, identical dimensions were chosen for the cylindrical tanks, in order to distribute the mass symmetrically. However, the position of the identical cylindrical tanks on the plate is not optimum, as there is a great deal of leftover space. This space could be used if the lengths of the cylinders were allowed to be different from each other; thus more propellant could be stored (see Figure 15-20). In the identical configuration, this loss of additional storage space in the length of the cylinders must be compensated by increasing the radius of the cylinders, which in turn raises the height of the spacecraft.

The lengths of the cylindrical tanks are limited by the size of the octagon mounting plate, as well as by the central spine that runs through the structure (see section 15.5.3).

Through basic analysis of the geometry of the mounting plate, it was determined that use of the bottom level is maximized by the combination of one pair of 42 cm-long cylinders and one pair of 33 cm-long cylinders (see Figure 15-20). For a given radius, the equivalent volume of four identically-sized spherical tanks requires around 7 cm of additional height (calculations were done for a mono-propellant, blow-down system supplying 450 m/s of ΔV). This difference in height is significant for a small spacecraft. Thus, cylindrical tanks of the size described above were chosen for Modsat.



4 identical cylindrical tank distribution   Optimum tank sizes and their distribution

Figure 15-20: Location and Size of Cylindrical Tanks

15.5.2.6.3 Bi-propellant versus Mono-propellant

Although bi-propellants have greater specific impulses, I_{sp}, than mono-propellants, Table 15-19 shows that the difference in required volume between bi-propellants and mono-propellants is small. This comparison was done by assuming a configuration of four cylindrical tanks, blow-down regulation, and enough propellant to deliver 450 m/s of ΔV. The bi-propellant system used two oxidizer and two fuel tanks. The fuels used in the comparison were those that deliver the highest specific impulse of the candidate blends listed in Table 15-16 and Table 15-17. Thus, the mono-propellant system used the 24% HN - 76%N_2H_4 blend (I_{sp}=263 m/s), while the bi-propellant system used the Fluorine - Hydrazine combination (I_{sp}=363 m/s).

Table 15-19: Mono-propellant and Bi-propellant Results

Propellant Type	42 cm Cylinder Radius	33 cm Cylinder Radius
Mono-propellant	9.91	9.96
Bi-propellant Fuel	8.00	8.92
Bi-propellant Oxidizer	8.04	8.97

Although there was not a big difference in the heights of the cylinders, the mono-propellant configuration was chosen as the Modsat standard because of its simplicity, cost, and reliability advantages (Sackheim and others, 1992:645).

15.5.2.6.4 Blow-down versus Regulated

Modsat does not require the capability for high-level long-duration thrust. In fact, it is clear that a low-thrust, small propulsion system is adequate. Thus, even though the allowable thrust of a blow-down propellant feed system decreases with burn time, it is suitable for Modsat. Moreover, since blow-down systems do not need regulators, and sometimes even filters, they are simple and reliable (Sutton, 1992:326). Therefore, the blow-down configuration is the best choice for the Modsat propellant feed function.

15.5.2.6.5 Pressure Range for the System

The mass and volume of the pressurant are dictated by the minimum pressure allowed for the propulsion subsystem. For a nominal propulsion system, end-of-life pressure is about 680 kPa (100 psia), and beginning-of-life pressure is about 2.07-2.76 MPa (300-400 psia) (Sackheim and others, 1992:654). For the propellant tank sizing calculations, initial pressure and final pressure were chosen to be 680 kPa - 2.76 MPa (100-400 psia), respectively.

15.5.2.6.6 Propulsion Structures and Materials

The propellant database used for this study covers only major hydrazine blends and combinations. Thus, the structures and materials must be compatible with hydrazine fuels. Information on the general subject of compatibility of materials with hydrazine fuels is extensive, and a complete discussion of the subject is beyond the scope of this study. However, a few guidelines with regard to compatibility can be suggested.

Hydrazine is compatible with a wide variety of the materials that could be used in the construction of the subsystem. However, care must be exercised in selecting suitable materials, due to the reactive properties of hydrazine and the necessity of minimizing propellant spills and leaks. Since hydrazine reacts with air, as well as some metal oxides and oxidizing agents, and absorbs water readily, storage and transport systems must be free from air, moisture, rust and contamination. Additionally, the lubricants, solvents and gaskets utilized in these systems must be chemically inert to hydrazine.

In a hydrazine system, those metallic components which may have long-term contact with the propellant include the propellant tank(s), propellant lines, and valves. For a system utilizing neat hydrazine (by far the most common), the list of compatible metals includes certain titaniums, stainless (corrosion resistant) steels and aluminums. If the selected propellant is a hydrazine and water blend, the aluminums are ruled out; if a blend utilizing hydrazine nitrate is used, the stainless steels are eliminated. This information is summarized in Table 15-20.

Table 15-20: Compatible Metals for Hydrazine Systems

Material	Neat Hydrazine	Binary N_2H_4 / H_2O Blends	Binary $N_2H_4/N_2H_5NO_3$ Blends	Ternary $H_2 H_4/H_2O/N_2 H_5NO_3$ Blends
Titanium (6A1-4V, pure)	Acceptable	Acceptabl e	Acceptable	Acceptable
Stainless Steel (302, 303, 304L, 316, 17-7, 430, 446, 355, 350)	Acceptable	Acceptabl e	Not Acceptable	Not Acceptable
Aluminum (6061-T6, 2014-T6)	Acceptable	Acceptabl e	Marginal	Not Acceptable

Source: Hydrazine Handbook 2-20.

The propellant which has the highest specific impulse (I_{sp}) is the 24%HN and 76% N_2H_4 blend. Only titanium is acceptable for that propellant. Therefore, titanium was chosen as the structural material, for its compatibility with all of the hydrazine blends.

15.5.2.6.7 Thrusters

The short mission life, weight and volume constraints point to the use of a single string thruster combination. Therefore, Modsat uses the minimum number of thrusters necessary to perform the intended propulsion functions. It was determined that six thrusters would be adequate. The two axial thrusters are placed vertically, perpendicular to the bottom plate. The roll thrusters are located on the sides, tangential to the edge of the octagon. The pitch thrusters are located at the sides, and are tilted at an angle as shown in Figure 15-21.

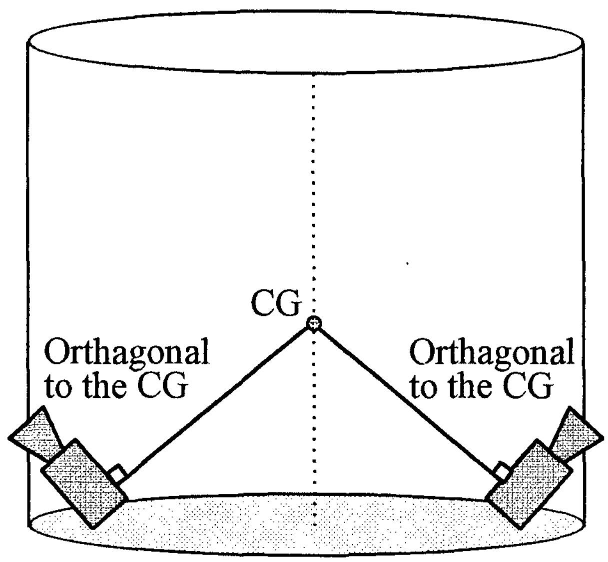


Figure 15-21: Location of Pitch Thrusters

The thrusters are located inside the bus, at the bottom level, and are located as far as possible from the center of gravity, in order to produce a large moment arm. The size of the thrusters is normally dictated by mission requirements. However, since Modsat must accommodate a variety of mission profiles, some of them requiring large amounts of thrust, a highly capable set of 22.4 N (5 lbf) thrusters was chosen. It should be noted that the Modsat computer model can easily be used to model the use of different thrusters, should the need arise.

15.5.2.7 Propulsion Alternatives

After making the design decisions discussed above, the remaining propulsion alterable is the amount of propellant, which dictates the size of the cylindrical tanks. Thus, all of the major system alternatives for this study were designed with a mono-propellant (24%HN and 76% N_2H_4 blend), blow-down pressure fed system with cylindrical tanks.

Although the mission requirements are uncertain, the idea behind the propellant alterable was to provide different amounts of ΔV capability. This capability must be divided among the spacecraft ΔV budget, which accounts for tactical maneuver capability, mission life time, retirement, and orbital accuracy. For a given amount of propellant, this budget must be managed by the user of Modsat.

The baseline orbit for Modsat was chosen to be 350 km. The amount of ΔV required to maintain this orbital altitude is 62.2 m/s per year. At this altitude, the amount of Δ V required to make a one degree plane change is 134.3 m/s (Larson and Wertz,1992: back cover). It was assumed that momentum management will require 35-50 m/s of ΔV per year. The chosen propulsion alternatives for this study are shown in Table 15-21.

Table 15-21: Propulsion System Alternatives

Alternative	Altitude Keeping (m/s)	Momentum Dumping (m/s)	Inclination Change (m/s)	Total Δ V (m/s)
1	62.2	37.80	------ / 0	100
2	62.2	36.35	201.45 / 1.5	300
3	62.2	52.05	335.75 / 2.5	450

Although the placement of thrusters was fixed, for each alternative the propellant tanks are located as near as possible to the center, in order to minimize the effect of diminishing propellant.

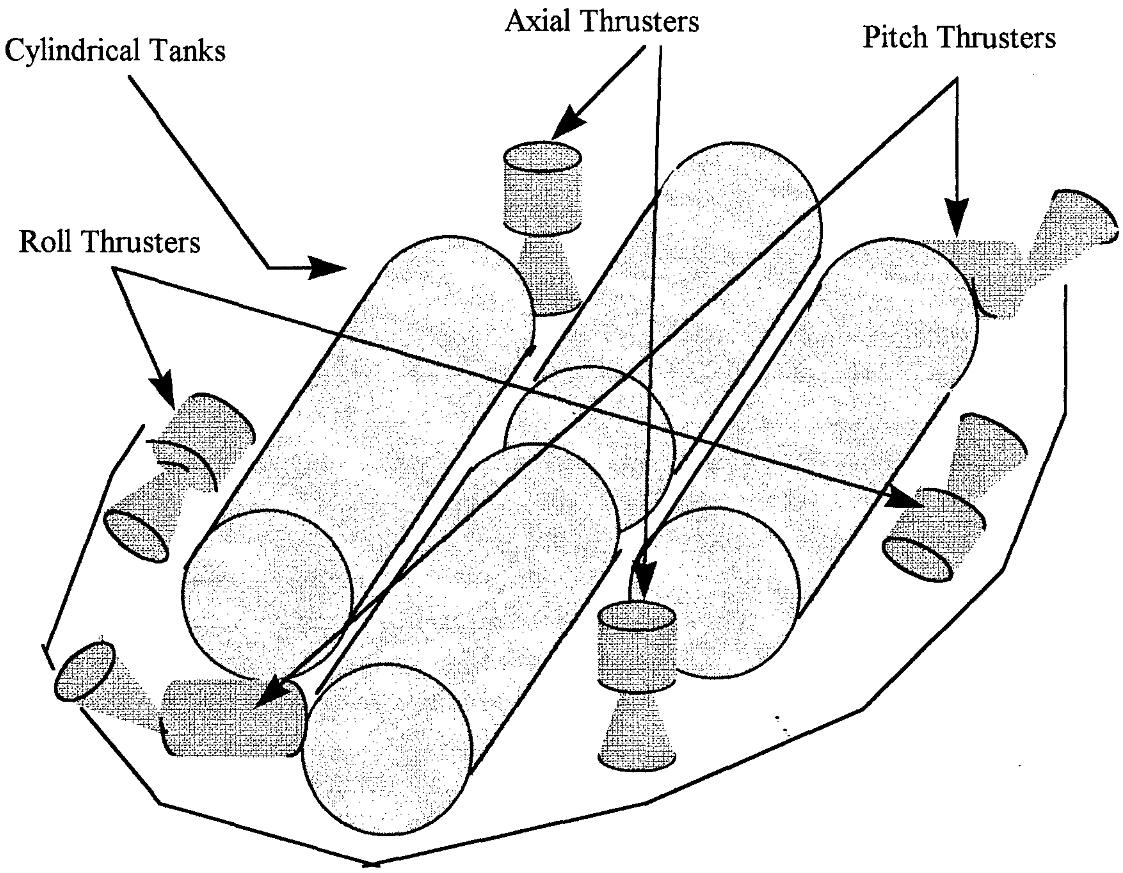


Figure 15-22: The Propulsion System

15.5.3 Structures and Mechanisms

15.5.3.1 Introduction

This section of the report presents a systems engineering approach to designing the structure of Modsat. This structural trade study details the steps and assumptions made to reduce the many possible structural design architectures. Only those viable alternatives satisfying the specific satellite missions called out for Modsat were considered for modeling. The best structural designs from the several possible Modsat designs are presented.

To conduct this structural study, a clear understanding of the problem was necessary. The problem is to design a small and inexpensive, light-weight structure capable of supporting tactical operations. To solve this problem, objectives must be defined. Among the many objectives and values for Modsat, low-cost and the desire to "launch-on-need" have the greatest influence on the structural design. Fulfilling these attributes is key if Modsat is to support various tactical operations anywhere in the world. Therefore, the satisfaction of these constraints was a major design goal for this structures design study.

15.5.3.2 Determining Feasible Structural Designs

In support of a low-cost design, graphite and composite materials were not considered. Although graphite-epoxy composites provide "...extremely high stiffness-to-weight ratios..." , they are costly and "...often require a long, expensive development program to establish manufacturing processes" (Doukas and others, 1992:435-441). Because the use of composites increase the complexity of the structure and drives up construction costs, the use of other metallic alloys, such as aluminum, is more desirable. According to Doukas, et al, "Aluminum is relatively light-weight, strong, readily available, easy to machine, and low in raw material cost" (Doukas and others, 1992:435-441). Therefore, only aluminum alloys were considered for the main structure of Modsat. In fact, until recently, the primary structure of MightySat, a common bus platform for testing new technologies for Phillips Laboratory, was being built exclusively with aluminum (MightySat TRD, 1996:79).

"Launch-on-need" implies the ability to launch within a few days, as opposed to several months, as with current launch schedules. Since the satellite bus must be able to support multiple mission modules, modular design schemes seem to be the most viable answer to a quick response launch. To incorporate modularity into the Modsat design, the team developed modular mounting plates. Such architectures allow for quick assembly, test, and check out. Of course, this will necessitate the development of flexible operating system software to compensate for hardware changes. To further satisfy the "launch-on-need" requirement, the satellite bus design must allow for quick mission module and launch vehicle integration. In addition, the ease of removing, replacing, and adding internal subassembly components is essential to last minute launch preparations.

To begin studying the structural design, the team first looked at the volume considerations governed by the payload shroud of the launch vehicle (see Figure 15-23). As shown, the team divided the total satellite into two main parts: mission module and satellite bus. The mission module is comprised of the mission payload and its supporting subcomponents. The satellite bus, containing the CPU, power, communications, etc., maintains the satellite and supports the mission module. For the purpose of the Modsat study, the team focused primarily on the satellite bus design, while striving to meet potential mission module requirements.

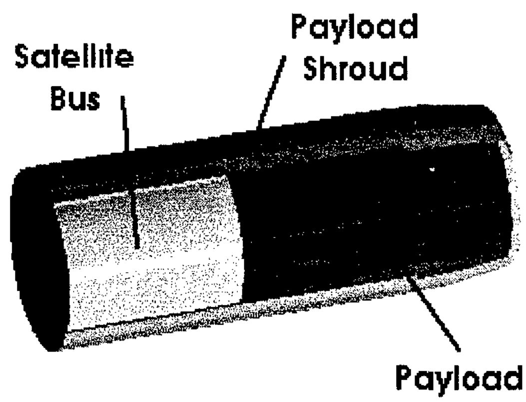


Figure 15-23: Payload Configuration

Since the Pegasus XL is Modsat's primary launch vehicle, the team first concentrated its efforts on ensuring that the satellite bus and mission module combination would fit within the Pegasus payload bay, as depicted earlier in this report.

The next consideration in selecting a satellite structure was to select a geometric shape that best utilizes the payload bay's volume and maximizes solar access (see Figure 15-24).

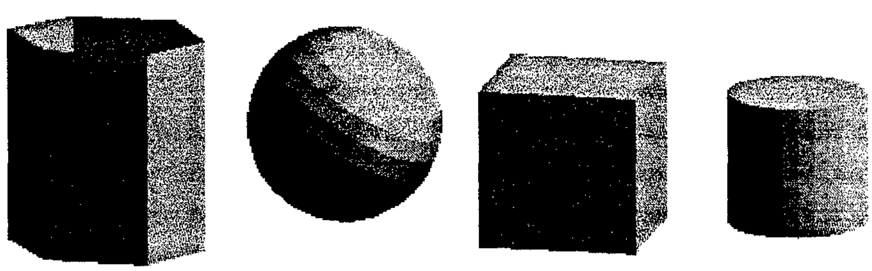


Figure 15-24: Structure Shapes

To do this, the team first maximized each geometric shape for volume within the cylindrical payload bay (34.27 inch height x 39.5 inch diameter) for the Pegasus XL 23" payload interface configuration. Once the shape's maximum dimensions were obtained, solar access was calculated, assuming the entire surface area of the geometric shape. The results shown below in Table 15-22 are taken from the calculations in Appendix B, Structures and Mechanisms.

Table 15-22: Geometric Shape Volume and Solar Access

Payload Bay (Cylinder) (34.37 inch diameter x 39.5 inch height)	**Volume (cm^3) Max: 688,176**	**Solar Access (cm^2) Max: 43,248**
Hexagon	569,117	39,276
Sphere	345,336	23,804
Cube	438,106	34,768
Cylinder	688,176	43,248

Although this table clearly shows the sphere and cube to be a poor choices for maximizing volume or solar access, the team discovered an interesting relationship. While performing the volume and surface area calculations for the various geometric shapes, the team found that except for the sphere, the geometric shapes are of similar construction. The remaining geometric shapes differing only in the number of sides, as depicted in Figure 15-25. Again referring to Figure 15-25 and Table 15-22, one can see how increasing the number of sides utilizes more volume and provides more solar access; however, there is a diminishing return to increasing the number of sides.

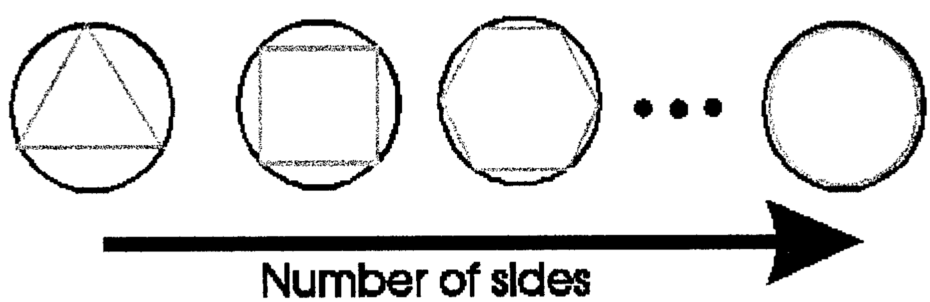


Figure 15-25: Geometric Shapes with Similar Properties

In fact, the volume and surface area calculations for the rectangle and hexagon were done with the same general equations. Although it appears that a cylindrical bus is the best geometric shape, the increased weight, limited access to subcomponents, and restricted solar wing placement in the stowed configuration make it a less desirable choice. Incidentally, Leritz and Palmer, in considering a geometric shape for their satellite design example, selected the cylinder and hexagon because they "distribute loads more uniformly" (Leritz and Palmer, 1995:490). From volume, surface area, and loading considerations, it seems likely that the best Modsat structural design is a "polysat," an n-sided polygon. By varying the number sides, the best Modsat design can be found. Therefore, the "polysat" design became the main focus of structural modeling.

Next, the team investigated what geometric shapes minimize materials and maximize internal subcomponent outside accessibility. Figure 15-26 below illustrates possible satellite bus designs.

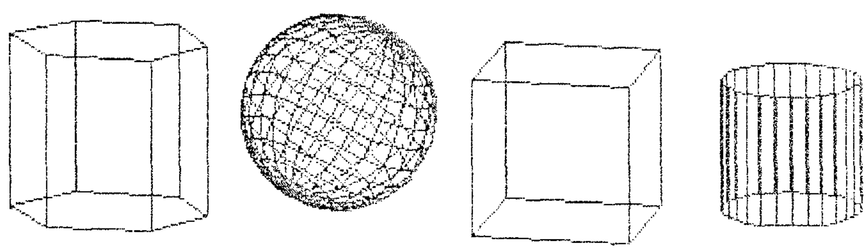


Figure 15-26: Exterior Structural Makeup for Geometric Shapes

In building a generic satellite bus, it is important to minimize weight while maximizing payload weight and meeting launch and on-orbit loading requirements. For example, if one considers the fact that about a fifth of the weight in communication satellites is framework (Larson and Wertz, 1992: 806), designing a satellite bus to meet total weight and loading requirements is difficult. The challenge is in satisfying these conflicting requirements.

Another consideration in the design of a satellite bus is the accessibility of internal subcomponents. This is concern with how easy it is to reach inside the satellite from the outside. This could be important for last minute launch preparations. Although it is possible to space out structural members farther apart for cylindrical and spherical bus designs, the number of members still exceeds that of the "polysat" designs. Also, internal component accessibility from the outside is more impeded than with "polysat" designs.

The final external structure consideration is the ease of mating the structure to the launch vehicle the mission module. Except for sphere all the designs have flat bottom and top surfaces, allowing for easier integration with the launch vehicle and mission module.

The results for material usage, component accessibility, and payload/launch vehicle interface compatibility are shown below in Table 15-23. Actual calculations for minimizing materials can be found in Appendix B. The component accessibility and payload compatibility ratings are based on 1-5 scale with 5 being the best. The following ratings are based on the team's assessment.

Table 15-23: Minimizing Materials and Maximizing Access for Geometric Shapes

Structure Type	Minimizes Materials (cm)	Component Accessibility	Payload/Launch Vehicle Interface Compatibility
Hexagon	1,124	4	5
Sphere	2,735	1	1
Cube	915	5	5
Cylinder	1,501	2	5

The next consideration of the team was to choose a satellite bus structure that best suits the goals for "low-cost" and "launch-on-need." Figure 15-27 below depicts just a few possible structural designs. The team evaluated each of these designs on how well they met the following objectives while supporting the "polysat" design architecture.

Modularity and Subcomponent Accessibility: How modular is the design and how easy is it to remove and replace subassembly components?

Materials Usage: Does this design minimize the use of materials in its construction?

Structural rigidity: What is the inherent strength and stiffness of the structure?

Manufacturing: What is the level of difficulty to construct the structure?

Based on these criteria the following designs in Figure 15-27 were evaluated using a pair-wise comparison technique (see Appendix B).

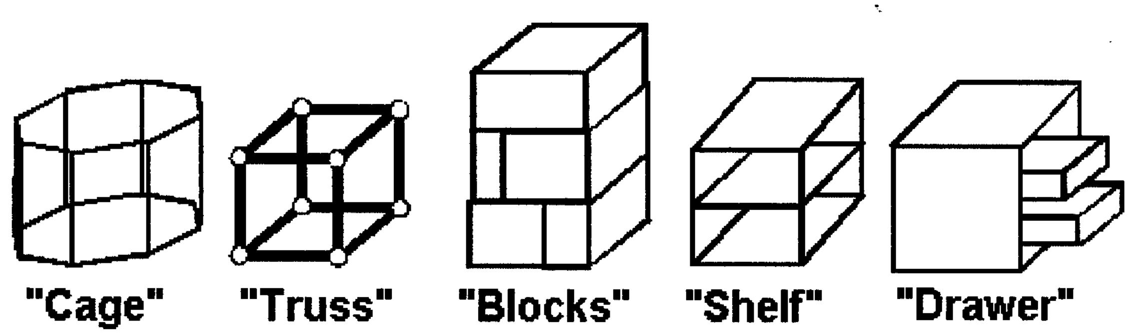


Figure 15-27: Satellite Bus Types

Cage: The structure is made up of a variable number of sides with each corner constructed with a load bearing beam. The beams used in the structure's construction are normally welded or bolted together or both. This design is the closet to the "polysat" design discussed earlier. Although this design received high marks for modularity and manufacturing, it scored below average in material usage and structural stiffness.

Truss: This structure mimics those used for future space constructions. The construction resembles that of tinker toys where beams are joined through the use of connecting hubs. This design scores above average in all categories except for modularity.

Blocks: This design incorporates the bolting together of subassembly components. The block design is the strongest of all design, but at the expense of added weight and machining.

Shelf: Most satellite designs use the box frame with shelves to layer the placement of payload and subassembly components. Once the shelf is installed, it becomes fixed with the structure. It becomes increasingly more difficult to remove as more components and shelves are added. This type of design received average scores for material usage and manufacturing, and below average scores for modularity and structural rigidity.

Drawer: This structure, a modified "Shelf" design, incorporates a pull-out and plug-in design. Although its modularity scored improved, the drawers add extra weight and complexity to the design.

Through pairwise comparison the team evaluated the five structural design for each objective listed above. Table 15-24 below shows how each satellite design did against the for each objective. For individual scoring results see Appendix B.

Table 15-24: Assessing Satellite Bus Types

	Modularity	Materials Usage	Structural Rigidity	Manufacturing	Total
Cage	13	9	5	14	**41**
Truss	4	14	10	11	**38**
Blocks	9	3	15	3	**30**
Shelf	5	9	3	8	**25**
Drawer	9	5	5	4	**23**

15.5.3.3 Structural Design Assessment

Up to this point in this study, two basic structural concepts were examined. First, the team started from a very basic level and investigated which geometric shape best maximizes the volume while keeping material usage down. Although it was determined that Modsat should have a "polysat" structure, the best polysat configuration can only be found upon modeling the complete Modsat bus design. Second, the team reviewed both current and revolutionary designs to determine the best functional design. Although the "Cage" and the "Truss" scored well, the "Truss" will require additional harnesses and brackets to support subcomponent attachments. Therefore, the team decided to design Modsat with the "Cage" configuration, which best supports the "polysat" design.

15.5.3.4 Polysat Design

Since it was determined that the best shape for Modsat is a "polysat", it is necessary to find the optimal number of sides to build the Modsat bus structure.

Although the number of sides is a design variable and can be specified in the model, this number should be minimized. By doing so, more space is allowed between support beams, granting greater side access to the subcomponents. However, the other consideration is the solar panels. As the number of sides increases, so does the number of folds in the solar array structure (see Section 15.5.3.5). This increases the complexity of the solar panel design and reduces the reliability of its structure. Therefore, the main driver is to select the minimum number of sides to obtain the solar panel area necessary to meet power requirements. Through extensive modeling the octagon was chosen as the baseline structure because it makes good use of the LV fairing volume while meeting power requirements.

15.5.3.5 Solar Panels Design and Assessment

The requirement for large amounts of electrical power played an important part in the design of the Modsat structure. To meet the power supply goal of up to 500 watts, considerably large solar panels are necessary. Modeling for the electrical power subsystem (EPS) determined that up to seven square meters of solar array area would be required (see section 15.5.6.4.3). Thus, the determination of the solar wing placement and deployment mechanisms proved to be a sizable challenge.

Since the structural design focused on maximizing internal volume, the team discounted the use of a folded-wing configuration for the stowed arrays, due to its inefficient use of space within the launch vehicle fairing shown in Figure 15-28.

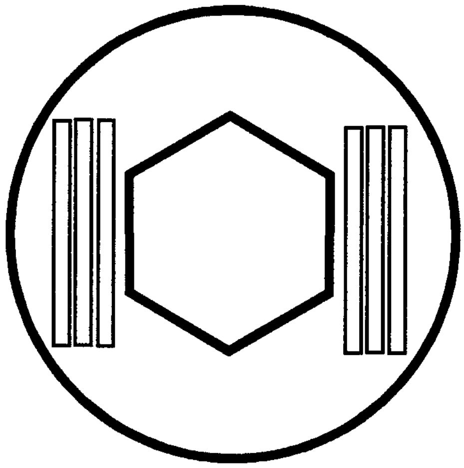


Figure 15-28: Folded Solar Array Configuration

The team also considered a deployment scheme similar to that used by Milstar, where the arrays spread out accordion-style from a small canister. Although this approach saves a great deal of space, it is very technologically complex and costly. Therefore, the team decided to wrap the solar array assemblies around the bus as shown below in Figure 15-29.

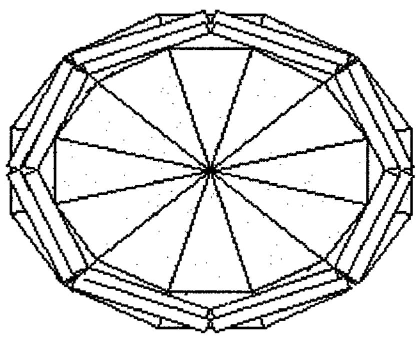


Figure 15-29: Top View of the Solar Panel Configuration During Launch

This approach was successfully used by the Clementine program (Clementine Report). In this configuration, the solar array assemblies must be constructed and hinged so as to fold around the polygon perimeter of the bus. This technique makes efficient use of the space within the launch vehicle fairing by placing the solar panels along the perimeter of the launch vehicle's payload bay as shown below in Figure 15-30. Moreover, since the most severe launch loads are in the axial direction, the vertical placement of the arrays has structural advantages.

Since the amount of power generated by an array is proportional to its area, it is desirable to maximize the surface area of the solar array assemblies. The surface area is determined by the circumference of the perimeter of the "polysat" and the height in the stowed configuration shown below.

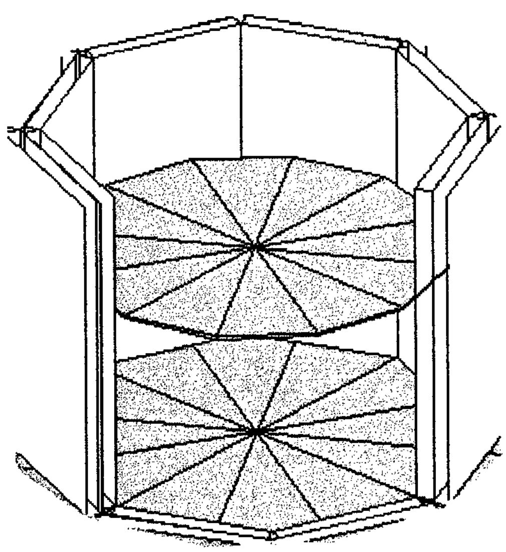


Figure 15-30: Side View of the Solar Panel Configuration During Launch

Note that in the stowed configuration, where the arrays are folded around the spacecraft structure, the height of the arrays corresponds to the width of the solar panels in the deployed configuration shown below.

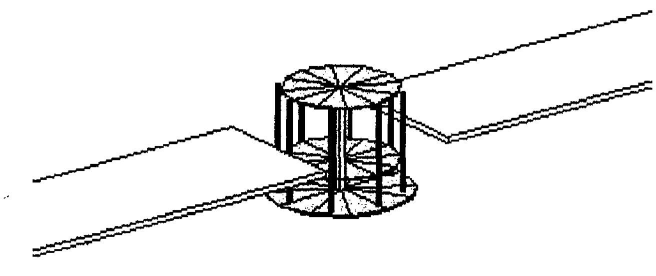
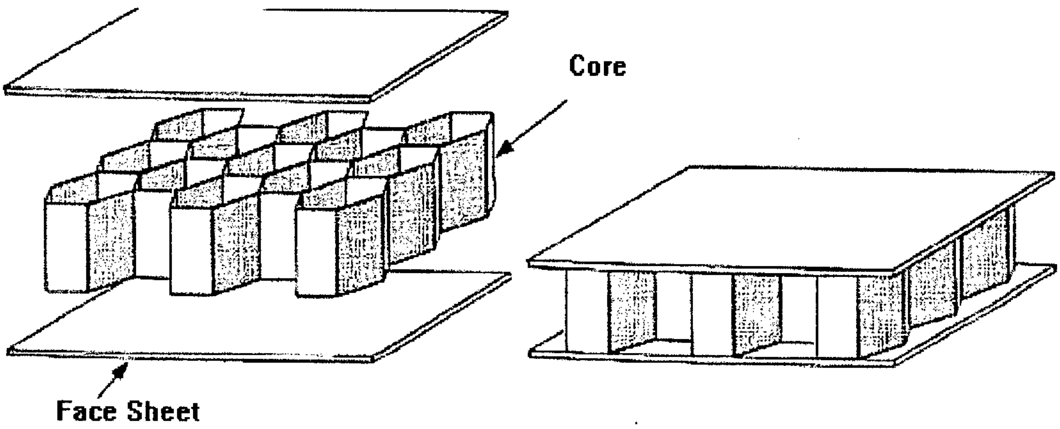


Figure 15-31: Deployed Solar Panels During On-orbit Operations

It was decided that in order to simplify the design and construction of the solar array assemblies, they would not be allowed to extend in height beyond the curve in the Pegasus fairing. Any protrusion beyond this curve would require the use of special hinges and mechanisms, which would drive up cost and decrease the structural reliability of the assemblies. Therefore, the maximum height of the stowed solar arrays is dictated by the distance from the LV-to-spacecraft interface plane to the curve in the fairing.

The Pegasus configuration that maximizes this distance is the 38" ring/without HAPS option. Throughout the modeling effort it became clear that the solar array surface area provided by one wrap around the bus would not be sufficient to meet the power requirement. Even with if the arrays could be perfectly folded around a cylindrical structure ("polysat" with infinite number of sides), the provided wrap area would still not be nearly enough to provide the required power. Although the number of wraps was left as a design variable for optimization, it was recognized that at least two wraps would be necessary. Due to deployment difficulties and reliability considerations solar wing wraps beyond two was not considered.

It must be noted that the solar array assemblies will have a certain thickness. Even though the solar cells are quite thin, they require a semi-rigid support structure such as the honeycomb panels shown below in Figure 15-32.



Source: Sarafin and others, 1995:528)

Figure 15-32: Honeycomb Panel Construction

This structure must include hinges. Moreover, since the solar array assemblies will be wrapped around the bus during the stressful environment of launch, they must include pads, bumpers, and other protection devices. The thickness of the assemblies can be varied in the model. To be conservative the team chose four centimeters as solar wing thickness. Thus, each wrap of the assemblies takes eight centimeters from the allowable diameter of the bus.

As discussed earlier, it is clear the wrap around design provides the best means of maximizing solar wing area with minimum volume impact for the mission module and satellite bus. As for multiple wraps, beyond two is a major design challenge and should be discouraged.

15.5.3.6 Structure analysis

In the evaluation of satellite structures, one must consider the loads on the satellite during preprocessing, launch, and on-orbit operations. The satellite loading environments also include storage and handling, transportation to the launch pad, and on-orbit thermal stresses. Because the most significant loads are during launch and ascent, the team only investigated axial and lateral loads. For the Pegasus launch vehicles these loads can be considerable (Everett, 1996). For the purpose of this study, the team used 13 g's axial loading and 6.0 g's lateral loading, as specified in Table 1 in "Tactical Support Satellite/Standard Payload Interface (TSS/SPI) Study" (Kim and Law, 1994:13). To ensure a structural integrity, the team built the design with a 1.6 yield design factor of safety for a "no structural test" specified in Table 11-48 of SMAD (Doukas and others, 1992:439).

To show the first order calculations for a satellite structure design, axial and lateral loading, natural frequency, and deformation analysis was performed on all seven of the satellite architectures for this study (see System Synthesis).

15.5.3.6.1 Axial and Lateral Bending Loads

The analysis of axial and lateral bending loads can be best visualized with Figure 15-33 below.

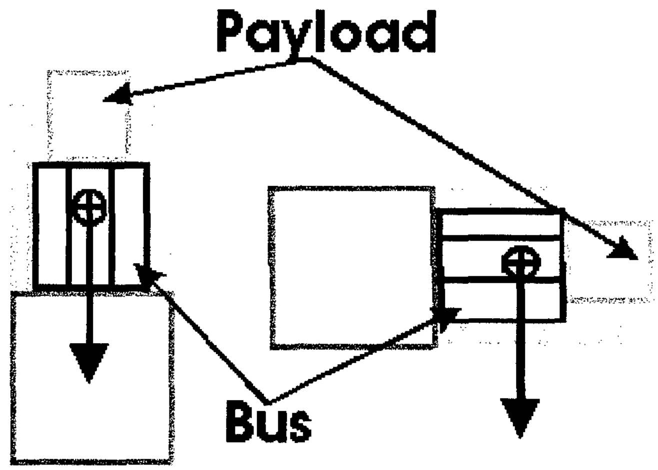


Figure 15-33: Axial and Lateral Loading on a Satellite Bus

To begin the structural analysis, the team investigated the optimal beam diameter and thickness. If one knows the critical buckling conditions (Pcr), the length of the beams (L), the number of sides, and the structure's material properties (E), one can calculate the area of inertia.

$$P_{cr} = \frac{\pi^2 EI}{4L^2} \qquad \textbf{(Eqn 15-17)}$$

From this one can model the beam's radius and thickness to obtain the lightest structure, satisfying the axial and lateral loading conditions (see Appendix B). Because each Modsat design will produce different loading conditions, these calculations give the designer a starting point.

With this analysis complete and some preliminary modeling, the team selected a octagon structure for all structural designs. Except for MIDTAC-23 all the remaining satellite designs were constructed with beam diameters of 4 cm and thickness of 1 cm.

Those parameters as well as the other key characteristics of the Modsat bus structure. are presented in Table 15-25 below.

Table 15-25: Structural Design Specifications

	Beam diameter (cm)	Beam thickness (cm)	Structure mass (Kg)	Structure height (cm)	Plate placement (cm)
MAXTAC	4	1	66.42	71	21,50
MIDTAC	4	1	65.75	68	18,47
LOWTAC	4	1	64.62	63	13,42
MAXTAC-N	4	1	65.97	69	21,63.8
MIDTAC-N	4	1	65.3	66	18,60.8
LOWTAC-N	4	1	64.17	61	13,55.8
MIDTAC-23	5	1.5	86.80	68	18,47

With these structural designs known the next step was to evaluate them for critical loading discussed above and bending stresses (σ_b) and equivalent axial loading (Peq) shown below:

$$\sigma_b = \frac{M \cdot c}{I} \qquad \textbf{(Eqn 15-18)}$$

$$P_{eq} = P_{cr} + \frac{2 \cdot M}{R} \qquad \textbf{(Eqn 15-19)}$$

Where

σ_b: Bending stress
M: Bending moment
c: Distance from neutral axis
I: Area of inertia
Peq: Equivalent axial load
Pcr: Critical buckling load
R: Radius of the satellite bus

Equivalent axial loading (Peq), which includes axial, lateral, and bending loads, is a measure of how much a loading a structure can withstand before buckling. Peq will be the primary measure for ensuring satellite structural integrity during launch.

Since these relationships are incorporated in the Modsat model, the model was used to evaluated each of the seven designs. All designs exceeded the structural limits of aluminum alloy 7076-T6 and the 494 pound loading conditions (at satellite cg). The results are shown below.

Table 15-26: Structural Results of the Modsat Designs

	Critical Load (Pcr) (1×10^4 N/m^2)	Bending stress (σ_b) (1×10^6 N/m^2)	Equivalent Load (Peq) (1×10^4 N/m^2)
MAXTAC	7.214	6.044	7.453
MIDTAC	7.885	5.825	8.111
LOWTAC	9.230	5.465	9.437
MAXTAC-N	7.651	6.037	7.877
MIDTAC-N	8.385	5.820	8.599
LOWTAC-N	9.867	5.495	10.060
MIDTAC-23	11.460	4.107	11.820
Aluminum 7076-T6	N/A	480	N/A
Max Load	4.576	N/A	4.576

15.5.3.6.2 Natural frequencies

A determination of the natural frequency is critical, in order to ensure that the structure will not resonate at a frequency equal to or less than the natural frequency of the launch vehicle. In this case, the natural frequency of the Pegasus is 18 Hz in the lateral and axial directions from Table 18-9 in SMAD (Loftus and Teixeira, 1992:688). The governing natural frequency equations from Figure 11-41 in SMAD are shown below (Doukas and others, 1992:454). Both equation are measured in Hertz (Hz).

$$\text{natural lateral frequency} = 0.276\sqrt{\frac{EI}{mL^3}} \qquad \textbf{(Eqn 15-20)}$$

$$\text{natural axial frequency} = 0.160\sqrt{\frac{AE}{mL}} \qquad \textbf{(Eqn 15-21)}$$

Where

E: Modulus of Elasticity
m: Mass per unit length
L: Length of beam
I: Area of inertia
A: Cross-sectional area

Again, the Modsat model was used to determine the natural frequency for all of the Modsat design. All exceeded the axial and lateral frequency for the Pegasus launch vehicle. The results are shown below in Table 15-27.

Table 15-27: Natural Structural Frequencies for the Modsat Designs

	Lateral Frequency (Hz)	Axial Frequency (Hz)
MAXTAC	19.81	403
MIDTAC	21.26	404.7
LOWTAC	24.11	407.7
MAXTAC-N	20.76	404.1
MIDTAC-N	22.33	405.9
LOWTAC-N	25.33	407.5
MIDTAC-23	19.16	476.8
Pegasus Launch Vehicle	18	18

15.5.3.7 Conclusions

This study has eliminated numerous satellite design alternatives. The best Modsat design is an octagon "polysat" constructed to fit on the Pegasus XL, 38"/without HAPS. The Modsat satellite design also includes mounting plates to enhance modularity and supportability of the mission module.

All of the Modsat designs in Table 15-25 exceeded the axial and lateral launch loading limits while satisfying the natural frequency requirements for the Pegasus XL. Although this study provided some preliminary structural results, further modeling is necessary to determine if these Modsat designs will meet other loading requirements not discussed.

15.5.4 Thermal Control

15.5.4.1 Introduction

Thermal design begins with defining its purpose. As the name implies, thermal design is concerned with constructing a "...*thermal-control subsystem* ... to maintain all elements of the spacecraft system within their temperature limits for all mission phases" (McMordie, 1992:409). In every satellite design, the designer must consider the thermal impacts due to the sun, the earth, and internal heat generation. Considering each of these thermal sources is important to ensuring the subcomponents within the mission module or satellite bus operate within their prescribed temperature ranges shown in Table 15-28.

Table 15-28: Typical Spacecraft Component Design Temperatures

Component or subsystem	Operating Temperature (degrees in C)	Survival Temperature (degrees in C)
Digital electronics	0 to 50	-20 to 70
Analog electronics	0 to 40	-20 to 70
Batteries	10 to 20	0 to 35
Infrared detectors	-269 to -173	-269 to 35
Solid-state particle detectors	-35 to 0	-35 to 35
Momentum wheels, motors, etc.	0 to 50	-20 to 70
Solar panels	-100 to 125	-100 to 125

Source: Wingate, 1994:433

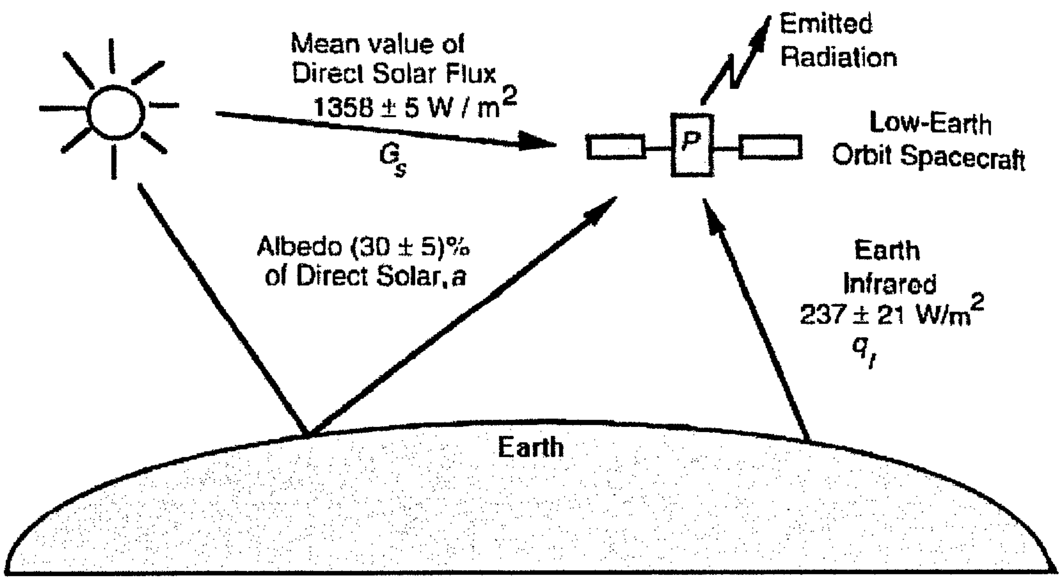
Just as the table suggests, the extent of thermal analysis depends on the thermal sensitivity of the satellite's subcomponents. As pointed out by McMordie, the power system has the greatest impact on the thermal design because of the electrical energy being dissipated throughout the satellite and the tight operating temperature limits of the batteries (McMordie, 1992:411).

15.5.4.2 Thermal Design and Modeling

Thermal design is not as trivial as it may seem. The earth, sun, and internal power thermal sources vary in duration and intensity with respect to the satellite. Modeling their thermal variances on the satellite throughout the satellite's orbit is difficult and beyond the scope of this thermal study. Instead, this thermal study will report the subcomponent thermal sensitivity results, which were used to design and develop the interface blanket for the satellite bus/mission module interface. Although this study will also report some preliminary thermal findings, only with a complete and more accurate thermal model can further research be conducted.

Thermal control can be accomplished either actively or passively, or both. The team decided from the start to maximize the use of passive systems to minimize cost, weight, and the power required of active systems (McMordie, 1992:413). Since "...preliminary mission design indicates that unmanned, low-Earth orbit spacecraft can be controlled passively", active systems were not investigated in this study (McMordie, 1992:413). Therefore, passive systems such as thermal coatings, thermal insulation, and space radiators are ideal devices for meeting thermal constraints in a satellite design (McMordie, 1992:411).

The team's analysis began with defining the thermal environment of Modsat. This environment, shown below in Figure 15-34, consists of four heat inputs/outputs.



Source: McMordie, 1992:410

Figure 15-34: Thermal-radiation Environment for a Typical Spacecraft

The heat energy shown by the arrows is transferred through radiation and is governed by the Stefan-Boltzmann equation shown below.

$$H = \sigma T^4 \qquad \textbf{(Eqn 15-22)}$$

H: Heat energy emitted
σ: Stefan-Boltzmann constant (5.67×10^{-8} Watts/m^2 K)^4
T: Temperature of the black body

The Stefan-Boltzmann equation does not account for conduction, the transfer of heat energy through physical contact between objects. Together, radiation and conduction considerations make up the extent of thermal analysis. Although thermal analysis may appear simple and straight forward, it is not. Complete thermal analysis of a satellite is complex because the designer has to consider all of the following thermal considerations:

- Subcomponents re-radiate energy to one another.
- Total eclipse times of satellite depends on orbit's size, inclination, and time of year.
- Subcomponents are made of different materials, exhibiting varying absorption (alpha) and emmissivity (epsilon) properties.
- Subcomponents have various geometric shapes, and may reside virtually anywhere within the satellite.
- Subcomponents generating heat may have varying on and off times, and they may have varying power levels when they are on.
- The attitude of the satellite (Wingate, 1994:433)
- Solar energy varies with earth's elliptical orbit about the sun (Wingate, 1994:440).

If a designer were to model and consider all the thermal conditions above, the thermal analysis would become very difficult, complex, and computer intensive. Thermal analysis essentially comes down to determining the number of conditions (variables) a designer wishes to track during the satellite's orbit. The designer must also consider how to model the satellite and its subcomponents. The designer could model the entire satellite as a box, a sphere, or a cylinder with some general properties, and obtain some overall satellite thermal approximations (McMordie, 1992:443-445,460-465). The designer could go a step further and model the temperature fluctuations for each subcomponent. The designer could take the final leap and use finite-element modeling, analyzing each subcomponent as a collection of smaller elements. As one author put it, "thermoelastic analysis with finite-element models ...is usually the best approach" (Lewis, 1992:305). Regardless of what modeling level the thermal designer decides to use, knowledge of the geometry of the satellite, the earth, and the sun is still the basis of thermal analysis. The designer should also keep in mind the "...geometry of a spacecraft is generally complex, leading to numerically complex analysis" (McMordie, 1992:409).

To develop a first order thermal model, the team needed to reduce the complexity of the thermal analysis. This started with making assumptions to eliminate a number of the variables already mentioned. Since satellite-earth-sun geometry (Qer) is the most challenging consideration, the team decided to consider only the sun (Qds) and earth (Qet) angles in the satellite's orbit. Because modeling the satellite's attitude throughout the orbit is a major undertaking, the team assumed the satellite's attitude was fixed in inertial space. The sun angle was modeled using the Beta (β) angle (see Eqn 1-23) for the entire orbit around the sun at 15 degree increments.

β = 23 degrees (earth's tilt) + orbit inclination (for the satellite) **(Eqn 15-23)**

The earth angle was assumed constant and always facing the top of the satellite. Although reflected solar radiation (Qer) is a major radiation contributor, amounting to 25-35% of direct sunlight, it was not considered in the thermal model (McMordie, 1992:410). Doing so would require extensive modeling of the sun-earth-satellite reflection angles throughout the satellite's orbit and the earth's orbit about the sun. With these assumptions in mind the team used equation 15-24 to perform the thermal analysis (Wingate, 1994:445):

$$Qint + Qds + \cancel{Qer} + Qet = \sigma \varepsilon AscTsc^4 \quad \textbf{(Eqn 15-24)}$$

Where:

- α: Solar absorptivity of the subcomponent
- ε: IR emittance of the subcomponent
- $\sigma = 5.67 \times 10^{-8}$ W-m^{-2}-K^{-4}: Stephan-Boltzmann constant
- Fet: Configuration factor
- Hsu: Solar constant
- Het: Earth's emitted IR
- PAS = Area vector*sun vector: Projected solar area
- Asc: Total surface area of component
- Qint: Heat output of the subcomponent
- Qds = α*PAS*Hsu: Incident solar energy on the satellite
- Qer: Reflected solar energy (sun-earth-satellite)
  - Note: Not considered in the Modsat model
- Qet = ε*Fet*Asc*Het: Earth emitted radiation
- Tsc: Temperature of subcomponent

Before calculating the amount of direct incident radiation on each subcomponent, the model converts the geometric shape of each subcomponent in the satellite into a box shape as shown below in Figure 15-35.

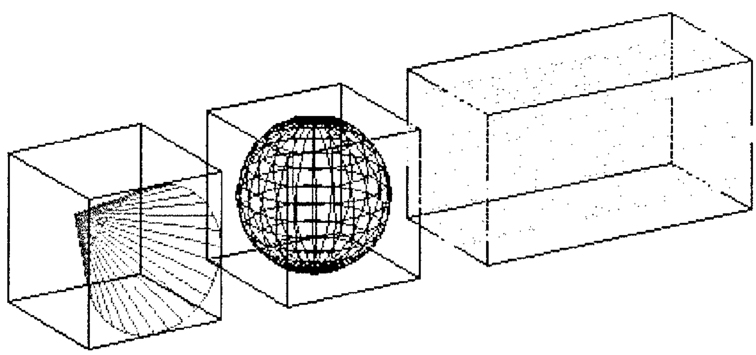


Figure 15-35: Conversion to Box Shapes by the Model

This technique within the thermal algorithm reduces the number of surfaces from several to six, but at the expense of introducing surface area and vector (facing direction) errors. Once the thermal model has determined the sun angles throughout the year plus one eclipse period, subcomponent temperature changes are calculated using equation 15-24. The resulting thermal information is then stored to be graphically displayed as shown in the next section.

Through thermal modeling, the team was also able to highlight those subcomponents more sensitive to thermal changes. The modeling analysis also showed that a majority of the subcomponents are sensitive to eclipse periods. Therefore, any follow-on thermal modeling should consider this phase of Modsat's orbit.

15.5.4.3 Thermal Properties of a Satellite Bus

To obtain a high level of understanding of these thermal effects on the Modsat structure, the team created a theoretical cylinder representing the height and diameter of MIDTAC. This cylinder, constructed of 7075-T6 mill aluminum, was run through the Modsat model with a 350 Km, 96.85 degree inclined sun-synchronous orbit.

Using the typical structures temperature of -45 to 65 °C (228 to 338 °K) from McMordie, the Modsat model calculated and graphed the temperature variations of the satellite bus throughout the year (McMordie, 1992:410). The red and blue lines depict the satellite's upper and lower temperature limits.

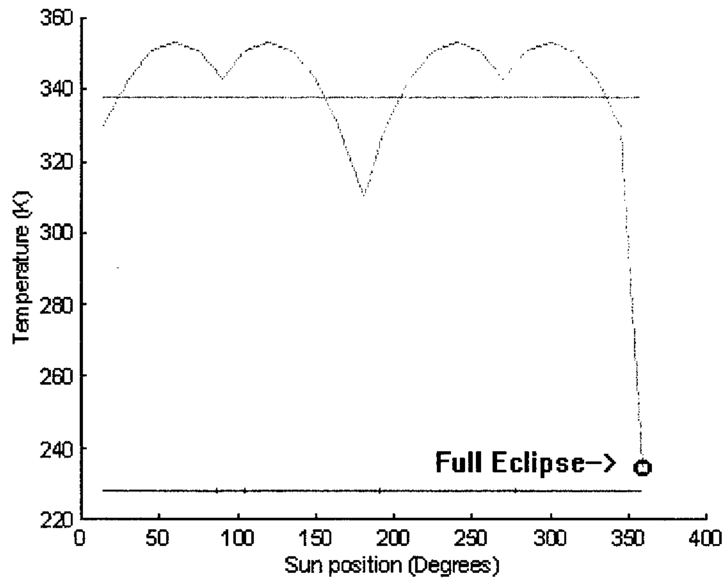


Figure 15-36: MIDTAC Bus Temperature Changes During the Year

This graph provided the team a rough approximation of the thermal conditions the hypothetical MIDTAC satellite bus could experience. Although the bus exceeds the upper operating limit, the data is based on a first order calculation and should not be accepted without further analysis. A more accurate model is necessary to complete the thermal analysis.

15.5.4.4 Designing Thermal Interface Plate Protection

To support the bus/mission module design discussed in the structure's trade study, the team designed a thermal blanket constructed of woven insulation to be placed directly below the top interface plate, as shown in Figure 15-37. This blanket (shown in green) will reduce the heat transfer between the bus (shown in light blue) and the mission module (shown in blue) by restricting temperature changes between the two structures.

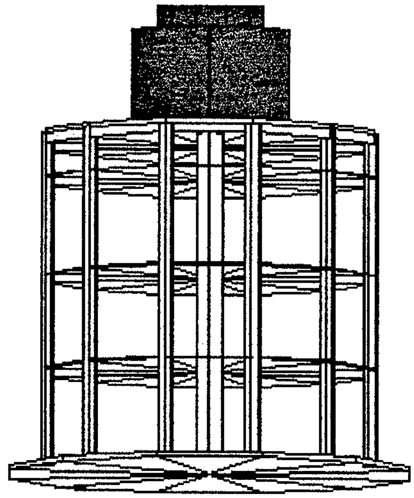


Figure 15-37: Thermal Protection of the Satellite Bus/Mission Module Interface

To restrict heat transfer between the bus and the mission module, the designer must apply the thermal conductivity equation shown below.

$$\dot{Q} = \frac{k_{aluminum} A}{L}(T_3 - T_2) = \frac{k_{insulation} A}{L}(T_2 - T_1) \qquad \textbf{(Eqn 15-25)}$$

Where

Q_dot: Heat flow
k: Thermal conductivity of materials
A: Area of heat flow
T: Temperature

With this equation, the thermal analysis between the mission module, the woven insulation, and the satellite bus can be conducted. The goal is to minimize the heat transferred between them. The $k_{aluminum}$ and $k_{insulation}$ terms represent the thermal conductivity of aluminum and woven insulation; the lower the number, the better the material will hinder heat flow. The area to which the heat transfers is designated A, and L is the distance the heat travels from one medium to another. To begin the analysis, one starts with T_1, the satellite bus temperature, and work backwards to obtain T_3, the temperature of the mission module. This preliminary analysis concentrated on the 100 °C differential of a typical bus structure as the baseline. Depending on the temperature required by the mission module, the minimum thickness of the woven insulation to preclude a 15 degree Celsius change between either the plate or the mission module can be calculated (see calculations in Appendix C, Thermal). For the Modsat the team designed for a 4 cm thick interface blanket.

15.5.4.5 Future Thermal Studies

The next generation thermal model should break the satellite into thousands of interconnecting nodes through some three-dimensional mapping scheme. Each node in the three-dimensional lattice would carry the attributes of position, local material properties, and associations with neighboring nodes. This idea comes partially from Wingate, who in fact suggested dividing up the satellite into 50 nodes, which are actually 50 separate surface areas making up the entire surface of the satellite (Wingate 1994:445-446). Each small piece would then be modeled for conduction and radiation thermal transfer.

15.5.4.6 Conclusion

Although the thermal model is far from being able to give the team an accurate representation of the thermal conditions the subcomponents will experience, it does highlight the level of sensitivity of the subcomponents. The first order thermal analysis also highlighted the need for further thermal modeling during eclipse periods.

The addition of a interface thermal blanket should prove useful for ensuring the heat flow between the satellite bus and mission module is kept to a minimum. However, the thickness of this blanket is contingent upon the thermal variations of the mission module and satellite bus. If the thermal variations between them are great, the interface thermal blanket may be too large taking up valuable volume.

Because many thermal design options are contingent on knowing the overall satellite configuration, thermal trade-off studies should continue throughout all phases of the satellite design.

15.5.5 Telemetry, Tracking, Commanding, Communications, and Data Handling

15.5.5.1 Communications System

The communications system for the satellite bus consists of the equipment necessary to relay commands to the vehicle from the ground and to send health, status, and in some cases, payload data, from the satellite to the ground. It is not uncommon for a satellite bus to employ two separate communications systems. One communications system nominally operates at a low data rate while the other operates at a high data rate. This configuration provides two different means of retrieving satellite data and commanding the vehicle. Operationally, the low data rate system is used to retrieve spacecraft health and status information and to send commands to the satellite bus. The Air Force Satellite Control Network provides this function to satellite programs such as Fleet Satellite Communications (FLTSATCOM), Defense Satellite Communications System (DSCS), Global Positioning System (GPS), and the Defense Support Program (DSP) (Muolo, 1993:178-183). The high data rate package is nominally used to receive mission module data and to send commands to the mission module. In contingency situations where one communications system is rendered inoperable, the alternate communications package can be used to receive spacecraft data and to command the vehicle. Vehicle equipment required to operate both systems is the same. This equipment includes an antenna, a diplexer (allows routing to and from the antenna), amplifiers, filters and demodulator/modulator.

Table 15-29 summarizes functions of the communications subsystem. It is adapted from Chapter 11.2 of Space Mission Analysis and Design, by Wertz and Larson (1992:368).

Table 15-29: Communications Subsystem

Specific Functions of the Communications Subsystem
Carrier Tracking - 2-way coherent communication (downlink is ratio of uplink frequency) - 2 way non-coherent communications - 1 way communications Command reception and detection - Acquire and track uplink carrier - Demodulate carrier and subcarrier - Derive bit timing and detect data bits - Resolve data-phase ambiguity if it exists - Forward command data, clock, and in-lock indicator to the subsystem for command and data handling

Specific Functions of the Communications Subsystem (continued)
<u>Telemetry modulation and transmission</u> - Receive telemetry data streams from the command and data handling subsystem or data storage device - Modulate downlink subcarrier and carrier with mission telemetry - Transmit composite telemetry and range signal to Earth or relay satellite <u>Ranging</u> - Detect and transmit ranging pseudorandom code or ranging tone signals - Retransmit either phase coherently or non-coherently <u>Subsystem operations</u> - Receive commands from the subsystem for command and data handling - Provide health and status telemetry to the C&DH subsystem - Perform antenna pointing for any antenna requiring beam steering - Perform mission sequence operations per stored software sequence - Autonomously select omni antenna when spacecraft attitude is lost -Autonomously detect faults and recover communications using stored software sequence

The conventional, or nominal, design of a command and data handling (C&DH) system requires both the command and telemetry equipment plus associated data lines be hard-wired together. The command decoder, telemetry format unit, and controller are usually contained within a single unit called the command and telemetry unit. The command decoder validates the command message and decodes the commands if the message is valid. The command decoder receives both bi-level commands (on/off commands) and serial (data) commands. Bi-level commands are sent to a command matrix, which is a board consisting of horizontal and vertical addresses. For a given horizontal and vertical address, a signal is routed to a subsystem to enable or disable a function. Serial commands are used for functions that are critical to the operation of the subsystem and possibly the satellite. Data is routed directly to a subsystem command buffer. That data is then read out in telemetry to verify that the data was loaded correctly before an executable command is sent.

Subsystem telemetry data is routed to the telemetry format unit. The analog data is usually converted to digital format before it is sampled, commutated with other digital data, and the result is placed into a telemetry masterframe. A telemetry masterframe contains all of the satellite data and consists of telemetry mainframes and subframes. Subframes contain small amounts of satellite data. These subframes are combined with other subframes to develop a telemetry mainframe of data. Telemetry mainframes are large amounts of satellite data that are combined with other mainframes to create a masterframe of telemetry data. The telemetry masterframe contains all of the satellite data.

All of these functions operate under the direction of the controller. The controller maintains the satellite timing and determines command priority if more than one commanding system is in use.

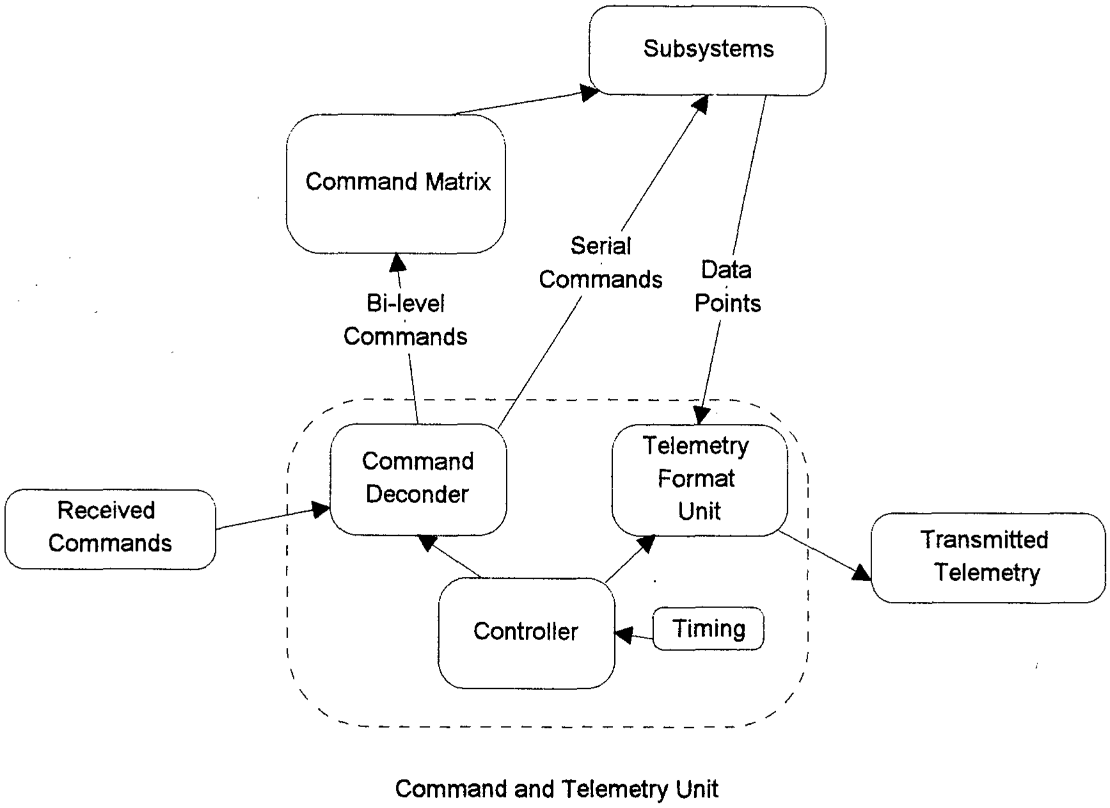


Figure 15-38: Nominal Style of Command and Data Handling

A possible Command and Data Handling architecture involves the use of a Satellite Operating System (SOS), which enables an extremely broad range of modularity. Using software, the main processor can perform the process of decryption/encryption, command controller functions, and data compression algorithms. Subsystem interfaces route through the Interrupt Request (IRQ) processor, which handles traffic deconfliction and prioritization. Instead of conventional hard-wired interfaces among subsystems, modular subsystems route their functional requests/needs through the central processor, which prioritizes subsystem requirements in view of overall satellite operations. Subsequently, appropriate action is directed to the proper subsystem in a format understood by the modular subsystem. In essence, SOS provides an interface to allow vastly different subsystems to communicate to each other and work together through a baseline structure. Further advantage is gained by the ability to change SOS on the fly while the satellite is in orbit, through the uploading of new software to the central processor. A block diagram is shown in Figure 15-39:

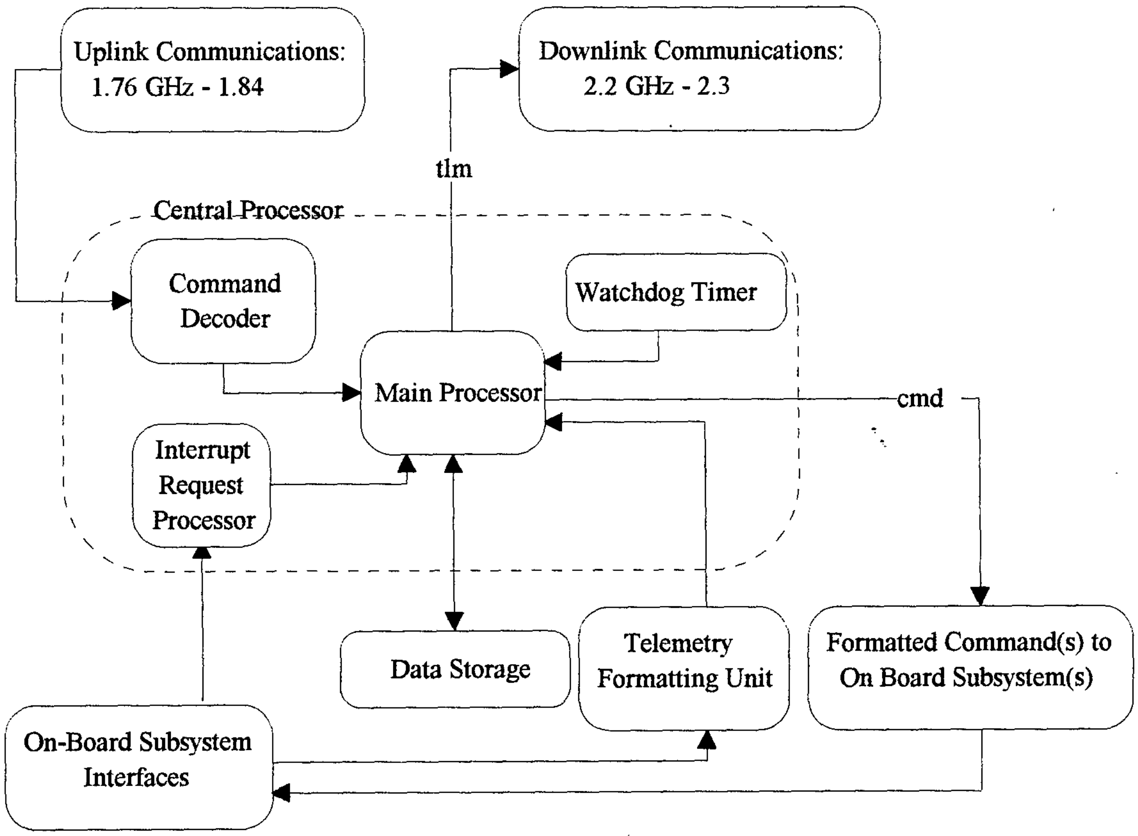


Figure 15-39: Proposed Style of Command and Data Handling

15.5.5.2 Subsystem Level Tradeoff

A subsystem design tradeoff was performed between the nominal C&DH design and the Satellite Operating System. The Satellite Operating System was selected as the baseline for the satellite bus design. This system was selected over the conventional means because the SOS offers more advantageous characteristics. The SOS will be a lighter-weight design because software can replace some of the subsystem hardware. For example, software code can replace the command matrix board, the encryptor, and the decryptor hardware. SOS offers yet another advantage. SOS has a means for modifying the downlink of health and status data in a more user friendly-manner. Software code can permit the formatting of satellite telemetry data in such a manner that telemetry data can be accessed by ground users in much the same way that users access the internet.

Another reason for the selection of SOS is that it is modular in nature. The modular characteristics of the system allow for easy addition or removal of components to the C&DH subsystem. By having IRQ checks, the system could operate with or without the additional components. If a component is not connected to the bus, the SOS would be aware of this status. If a component is added, "driver" software would be resident within the SOS program that checks to see if the added component is available for operation. Another advantage of selecting SOS is that the C&DH code could be modified while the spacecraft is in flight. While some spacecraft permit Attitude Control Subsystem modification, C&DH subsystem modification is not an option on most current satellite designs.

15.5.5.3 Data Handling

Instead of using conventional methods for data uplink/downlink, SOS can take advantage of data formatted in Hyper Text Markup Language (HTML), and more recently JAVA, which are standardized protocols for handling multimedia over the internet. This approach allows a user in a combat theater, using a laptop connected to a small, portable communications dish, to receive data from the satellite. A forerunner to this type of ground station is resident at the Naval Post Graduate School in Monterey, California (Bible, 1995). The user could review the data at will with an internet browser and be empowered to act more efficiently due to timely information flow. If the satellite is overhead and thus able to communicate for five to ten minutes during a pass, data can be dumped to a CD ROM. The user can alter the amount of data received to the level needed, providing a "data-on-demand" flow instead of working with the full amount of data downlinked from the satellite. Use of "data-on-demand" leads to more efficient use of communications bandwidth. Security plug-ins to the browser will handle Department of Defense encryption needs.

In the event a satellite has served its purpose and is no longer needed by personnel in a given military theater, security plug-ins can be deactivated on the fly, and the satellite can serve commercial purposes or even other military needs. This method of data handling enables the satellite to adapt to changing needs to suit multiple purposes, which means greater efficiency and far better economical use of the satellite than current methods.

In essence, the paradigm of a satellite as another internet node has far reaching implications. Various payload imagery and communications can be used rather efficiently by a user in the field with the point-and-click ease of a web browser. Satellite intelligence can be displayed in a user friendly format in near real time along with other multimedia, such as a voice message attached to the most recent imagery, giving the field commander in another remote location the user's perspective of a certain objective. Furthermore, the user can load scheduling plans for any connectivity desired with the satellite by uplinking desired parameters to the satellite as it passes overhead, or even via a "bent pipe" analogy (Klements, 1992) when the satellite is not in view. Range scheduling is thus coordinated between user and satellite instead of through traditional means.

15.5.5.4 Subsystem Interfaces

The Command and Data Handling subsystem interfaces with multiple subsystems at varying data rates; refer to Figure 15-40. It must handle payload telemetry and commanding, and provide 'store and forward' capability. Sensor information gives the central processor positioning data, which is then correlated with attitude control (propulsion subsystem). C&DH must be able to interpret command formats from the various subsystems, reply to subsystems in the appropriate format, as well as format telemetry and encrypt as necessary for downlink to ground stations.

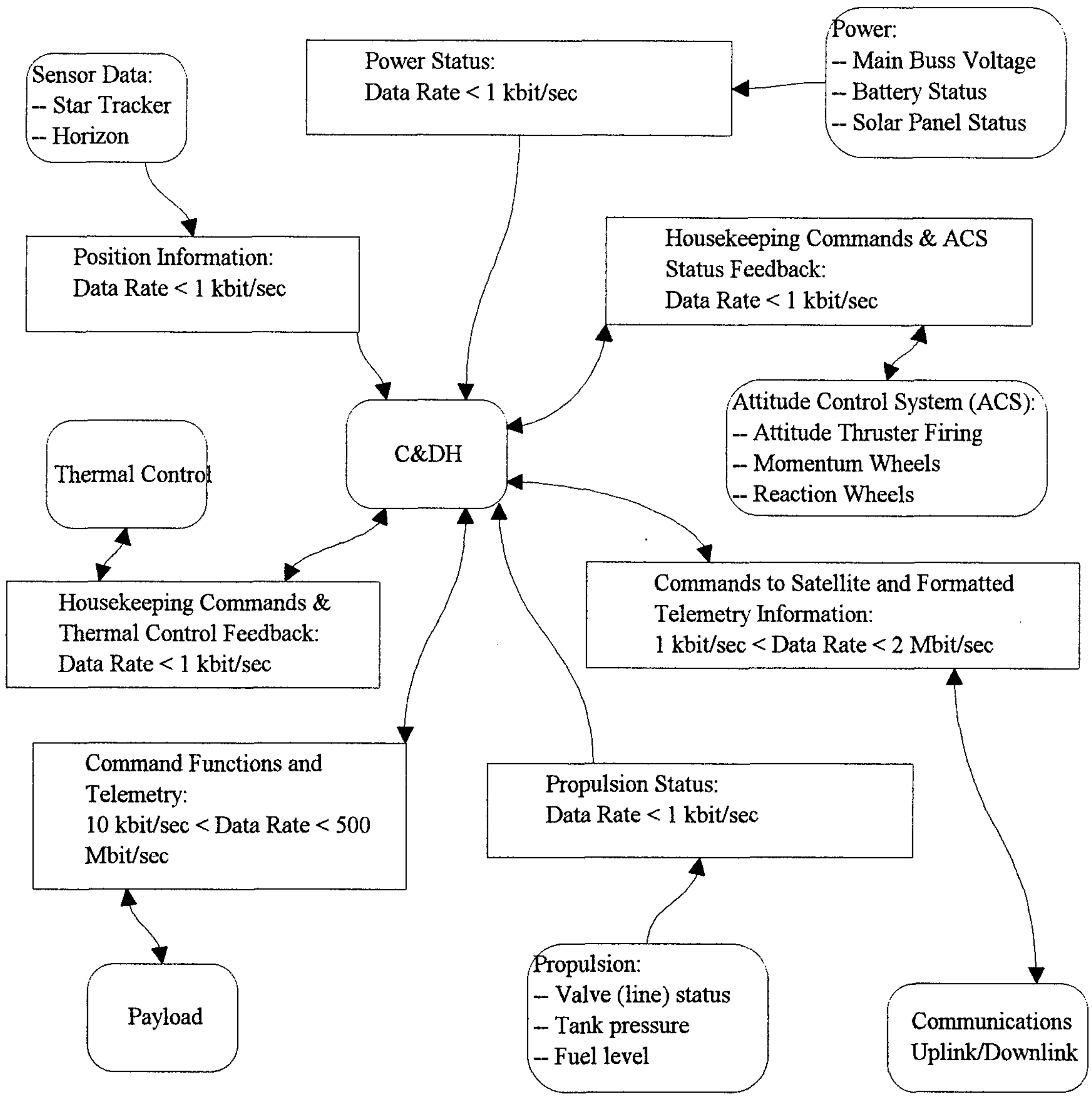


Figure 15-40: Command and Data Handling Interactions

15.5.5.5 Design and Sizing

15.5.5.5.1 C&DH Design Process

To design a satellite bus Command and Data Handling subsystem, the designer must understand the operation of the subsystem and consider many different aspects affecting the subsystem. The designer must also understand the effects these different aspects have on mass, cost, and power of the components used in the construction of the subsystem. Each of the following are drivers for determining subsystem mass, cost, and power consumption of the C&DH subsystem:

- Data rates are required for interface communications between subsystems, which also drives required clock speed and processor power.
- Crosslinking to other satellites drives the need for antenna, power, and computer processor capability.
- Data storage for mission modules drives the need for on-board computer memory.
- The available time for data downlink drives data rate requirements, which also drives the need of more computer processing power.

Other issues affect the means for downlinking mission module data to the ground station or other user, i.e., the communications aspect of the C&DH subsystem. These issues also affect mass, cost, and power requirements. A few areas of concerns are listed below:

- Minimum acceptable signal to noise ratio (SNR) drives power requirements for the data downlink.
- Minimum acceptable SNR for received signals drives the gain needed on-board the satellite.

Using this information as a background for starting the design, it is necessary that the subsystem engineer understand the mission of the satellite. Additionally, communication with other subsystem experts is also required. This interaction ensures that subsystem interface requirements are met and that C&DH system is designed correctly to perform its intended mission.

There are many issues affecting the design of a Command and Data Handling subsystem. Table 15-30 provides an outline for the steps involved in this process.

Table 15-30: Steps for Designing C&DH System

Step	What's Involved
[a] Obtain data rates for payload, subsystem interfaces, and communication uplink/downlink and crosslink	Receive information on mission module; get information on satellite bus; understand communication frequencies, bandwidths, and secure transmission requirements
[b] Prepare command list and telemetry list	Understand minimum telemetry points for satellite health, and interfaces needed to provide commands to satellite subsystems
Both of which yield	
[c] Processor power and memory requirements	Understand minimum processor power to handle maximum data rates for communication and housekeeping, along with minimum memory capacity for store and forward
Which yields	
[d] Overall processing requirements	Examine need for encryption, decryption, sequencing, and processing of commands and telemetry
Which leads to	
[e] Selection of equipment	Understand Commercial Off the Shelf (COTS) equipment, cost, size, mass, and Mean Time Between Failures (MTBF)

Step	What's Involved
If Mission Module Data is to be Downloaded:	
[f] Obtain bandwidth and carrier frequencies for uplink/downlink, along with acceptable SNR and access scheme	Establish power budget for communications and understand necessary footprint
Which yields	
[g] Data for link budget	Estimate atmospheric attenuation, noise, and interference. Perform link budget calculations.
Which leads to	
[h] Sizing communication system	Select antenna configuration, calculate size, and estimate mass. Estimate transmitter mass and power. Obtain cost figures.

Table 15-31 provides a parametric estimation for mass, power and size of a C&DH subsystem (Wertz and Larson, 1992:390). Similarly, data has been collected on the cost and reliability factors for C&DH subsystems. This information is displayed in Table 15-32 (Wertz and Larson, 1992:389).

Table 15-31: Estimation of C&DH Size, Weight, and Power

		Simple	Typical	Complex
Size (cm^3)	Command only	1500 - 3000	2000 - 4000	5000 - 6000
	Telemetry only	1500 - 3000	4000 - 6000	9000 - 10000
	Combined systems	2500 - 6000	6000 - 9000	13000 - 15000
Weight (kg)	Command only	1.5 - 2.5	1.5 - 3.0	4.0 - 5.0
	Telemetry only	1.5 - 2.5	2.5 - 4.0	6.5 - 7.5
	Combined systems	2.75 - 5.5	4.5 - 6.5	9.5 - 10.5
Nominal Power (watt)	Command only	2	2	2
	Telemetry only	5 - 10	10 - 16	13 - 20
	Combined systems	7 - 12	13 - 18	15 - 25

Table 15-32: Reliability and Cost Data for C&DH Subsystem

Requirement/ Constraint	Complexity Simple	Typical	Complex
Reliability- Class B Parts			
Single String	0.8233	0.7610	0.6983
Redundant	0.9875	0.9736	0.9496
Reliability- Class S Parts			
Single String	0.9394	0.9083	0.8285
Redundant	0.9987	0.9964	0.9829
Costs			
Class B Parts	100%	100%	100%
Class S Parts	400-500%	400-500%	400-500%

Modeling software has been developed to take advantage of the information presented in Table 15-31. Through the use of graphical user interfaces, the user has the option of designing a subsystem by its parts (command only and telemetry only) or as an integrated system. The user can also specify the desired level of complexity to design into the satellite bus. The code allows the user to enter the known information about the subsystem design. The unknown parameters are then calculated using an interpolation method. For instance, if the designer enters the weight of a command component, the corresponding size and required power are calculated and displayed.

15.5.5.6 Component Selection

One of the goals of the study was to use commercial off the shelf (COTS) components where possible. Information had to be gathered on the commercially available products before component selection could begin. Two of the companies that responded to our information request were Cincinnati Electronics and Aydin Vector.

These companies specialize in components compatible to both the Air Force Satellite Control Network's Space Ground Link System and NASA's Ground-Space Tracking and Data Network. Another company that responded was Honeywell. Each company sent a catalog describing the performance and physical characteristics of their products. It should be mentioned that the team was not able to gather component price information from any of the vendors.

Components from other satellite programs were also considered. The Jet Propulsion Laboratory (JPL) provided the team with a Flight Hardware Survey that included a database of the components used on JPL programs. The Clementine Program was also examined for Command and Data Handling components. The Clementine Program incorporated products considered to be state of the art. The benefit of using Clementine's products is that these components are generally smaller and lighter, and provided better performance than other commercial off the shelf products.

The Command and Data Handling components chosen for the generic satellite bus design were primarily rated on their performance-to-mass ratios. Cost was not a driving factor because component prices were not available. Discussions with satellite designers and engineers such as Richard Warner of AeroAstro, Dave Everett of NASA/Goddard Space Flight Center, and Joel Hagan of Phillips Laboratory's MightySat Program, revealed that Command and Data Handling is one of the most critical subsystems on-board the satellite. Without the proper functioning of this subsystem, the mission of the satellite would be lost. With this in mind, it was decided that only space-rated products would be used in the design of this subsystem.

Other factors were considered in the selection of components. A requirement for the transceiver and antenna was that they must be compatible with the Space Ground Link System. A small Cincinnati Electronics transceiver, component number TTC-307/702, was selected for the team's satellite bus design because the device operated using the Air Force Satellite Control Network frequencies. The amount of memory provided by a data storage device was also an important factor since the satellite needs the capacity to store both mission module data and state of health data. The team selected the Clementine Program's data storage device because it provided 2 Gigabytes of data for a mass of only 3.4 kilograms. Other data storage devices considered were much more massive. Realizing that a Satellite Operating System architecture would require strong computing power, the team chose a radiation hardened, 32-bit, high speed Reduced Instruction Set Computer (RISC) architecture employing Very High Speed Integrated Circuit (VHSIC) technology. This computer was developed by Honeywell for the Ballistic Missile Defense Office. The associated controller and memory modules were developed under sponsorship of the Naval Research Laboratory. The computer and its associated controller were light weight and extremely powerful.

Specific components were not selected for the high data rate communications system. As part of the design study, it was decided that the mission module developers would provide a transceiver and antenna for the satellite bus. This arrangement would provide more flexibility for the mission module. Interface data, power, mass, and size requirements would be provided to the mission module developers to assure compatibility with the Satellite Operating System. In the concept of operations, the use of the Satellite

Operating System would allow easier integration of a high data rate communications system.

15.5.6 Electrical Power

This section of the report discusses the design of the electrical power subsystem (EPS) for Modsat. The following areas are discussed:

- Problem Definition
- Objectives
- Functional Allocation
- Component Design
    - -- System synthesis
    - -- Tradeoffs and Decisions
    - -- Design Process
- Future Efforts

15.5.6.1 Problem Definition

The EPS must perform the following functions (McDermott, 1992:391):

- Supply a continuous source of power to all spacecraft loads
- Control and distribute power to the spacecraft
- Support power requirements for average and peak loads
- Provide converters for voltage, as required
- Provide command and telemetry capability for the EPS
- Protect the spacecraft against failures within the EPS
- Suppress transient bus voltages and protect the subsystem against faults
- Provide for switching to redundant spacecraft components

The EPS may also provide for reconditioning the batteries, if this is necessary.

Many elements of the spacecraft and its mission affect the design of the EPS. Among them are (McDermott, 1992:392):

- Average electrical power requirement

  Sizes the power generation system and possibly the energy-storage system

- Peak electrical power requirement

  Sizes the energy storage system and the power-processing and distribution equipment

- Spacecraft design life

  A longer mission life implies extra redundancy, as well as larger batteries and arrays

- Orbital elements

  Defines incident solar energy, eclipse/Sun periods, and the radiation environment

- Spacecraft configuration

  Affects the geometry of the power generation system

15.5.6.2 Objectives

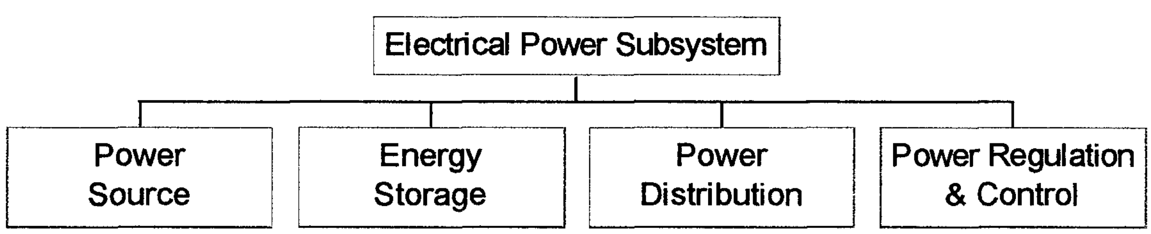
As with the other subsystems, minimizing cost is a key objective in the design of the EPS. An emphasis was placed on using readily available, relatively inexpensive components. Another key objective is minimizing technological risk. All spacecraft require power, and advancements in space power technology are continually made. But in light of the objectives, the EPS design emphasized the use of proven components and architectures.

A third objective which had a large impact on EPS design is maximizing flexibility. An effort was made to accommodate modularity and ease of integration, since each Modsat spacecraft could be a different configuration.

But of course, the overriding objective for EPS design is to provide sufficient power for the possible range of mission modules. The conclusion one must draw for the generic bus is that its EPS accommodate the most power-hungry applications planned to be flown on Modsat. Thus, a large amount of power must be available for the mission modules. However, not all configurations will require all the available power. A sound systems engineering approach must consider the impact of excess power (thermal effects, inefficient use of weight, etc.).

15.5.6.3 Functional Allocation

The EPS consists of four major functional areas, as shown in Figure 15-41: power source; energy storage; power distribution; and power regulation and control.



Source: McDermott, 1992:391

Figure 15-41: Electrical Power Subsystem Functions

The systems engineering process must occur in each area, to select a design that optimally satisfies the requirements and objectives. A simple block diagram of the EPS is shown in Figure 15-42.

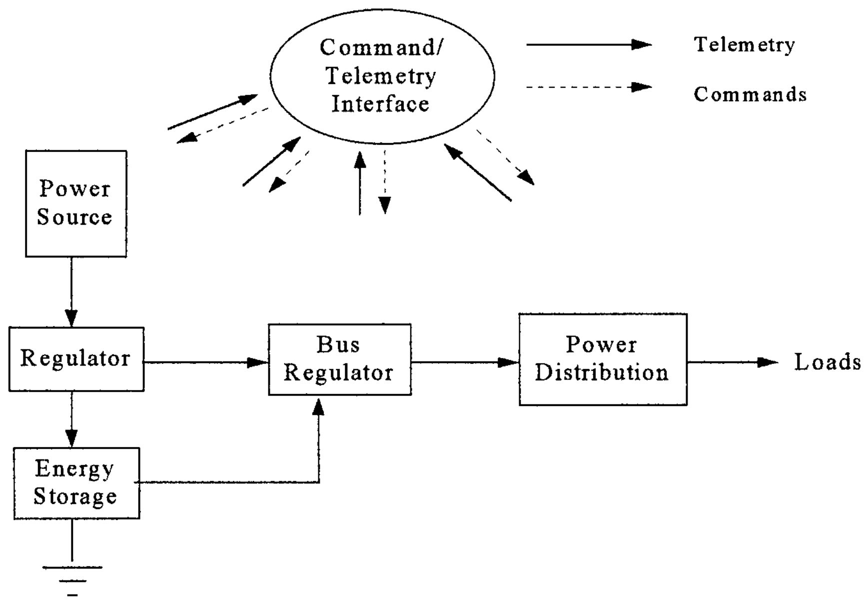


Figure 15-42: EPS Block Diagram

15.5.6.4 Power Source

The power source generates electrical power for the spacecraft, supporting the electrical loads over many orbits.

15.5.6.4.1 System Synthesis

Spacecraft typically use one of three types of power sources:

1. *Photovoltaic solar cells* are the most common power source for Earth-orbiting spacecraft. They convert incident solar energy directly to electrical energy (McDermott, 1992:392).

2. *Static* power sources use a heat source for direct thermal-to-electric conversion. Typical sources are plutonium-238 or a uranium-235 nuclear reactor (McDermott, 1992:392-393).

- The thermoelectric couple uses the temperature gradient from the slow decay of a radioactive source to provide the desired dc electrical output (McDermott, 1992:393).

- Thermionic energy conversion "produces electricity through a hot electrode (emitter) facing a cooler electrode (collector) inside a sealed container that typically contains an ionized gas" (McDermott, 1992:393). Electrodes evaporate at the emitter, flow across the gap, condense at the collector, and return to the emitter through the electrical load. Thermionic sources usually require the heat temperature of a nuclear reactor source (McDermott, 1992:394).

3. *Dynamic* power sources "use a heat source and a heat exchanger to drive an engine in a thermodynamic power cycle. The heat source can be concentrated solar energy, radioisotopes, or a controlled nuclear-fission reaction. Heat from the source transfers to a working fluid, which drives an energy-conversion heat engine" (McDermott, 1992:394). Radioisotopes and nuclear reactors provide continuous heat, and therefore do not require energy storage for eclipse periods. However, dynamic solar power sources retain the balance of energy as latent heat in a heat exchanger. This provides continuous energy to the thermodynamic cycle during eclipse periods (McDermott, 1992:394).

The most common of these sources is compared in Table 15-33.

Table 15-33: Power Source Comparison

EPS Design Parameters	Solar Photovoltaic	Solar Thermal Dynamic	Radio-isotope	Nuclear Reactor
Power range (kW)	0.2 - 25	1 - 300	0.2 - 10	25 - 100
Specific power (W/kg)	26 - 100	9 - 15	8 - 10	15 - 22
Specific cost ($/W)	2500 - 3000	800 -1200	16k - 18k	400 - 700
Hardness				
- Natural radiation	Medium	High	Very high	Very high
- Nuclear threat	Medium	High	Very high	Very high
- Laser threat	Medium	High	Very high	Very high
- Pellets	Low	Medium	Very high	Very high
Stability/maneuverability	Low	Medium	Low	Low
Degradation over life	Medium	Medium	Low	Low
Storage required for solar eclipse	Yes	Yes	No	No
Sensitivity to Sun angle	Medium	High	None	None
Sensitivity to spacecraft shadowing	Low (with bypass diodes)	High	None	None
Obstruction of spacecraft viewing	High	High	Low	Medium (due to radiator)
Fuel availability	Unlimited	Unlimited	Very low	Very low
Safety analysis reporting	Minimal	Minimal	Routine	Extensive
IR signature	Low	Medium	Medium	High
Principal applications	Earth-orbiting spacecraft	Interplanetary	Interplanetary, Earth-orbiting spacecraft	Interplanetary

Source: McDermott, 1992:393

15.5.6.4.2 Tradeoffs and Decisions

Solar photovoltaics are the typical power source for LEO spacecraft. They are proven and widely available. Solar power is adequate for the intended mission, and lighter (in terms of specific power) than the other alternatives. Although there are cheaper sources, i.e., solar thermal dynamic and nuclear reaction, these alternatives require more mass. Nuclear reactors, furthermore, require a fuel that has limited availability, while solar energy is unlimited. Reactors are appropriate for applications requiring a large amount of power, but not for the loads and environment expected for a small, tactical LEO spacecraft such as Modsat. The same can be said of the solar thermal dynamic alternative. The radioisotope alternative fares very well in all categories shown in Table 15-33 but one, wherein it is very cost prohibitive. Given the strong emphasis on affordable access to LEO missions, the radioisotope alternative was ruled out.

Thus, the power generation system of choice for Modsat is photovoltaic solar cells. These cells must be mounted on solar panels. Solar panels can either be planar or cylindrical. For planar panels, the "power output is proportional to the projection of their area toward the incident sunlight. Three-axis-stabilized spacecraft normally use planar arrays" (Reeves, 1992:317). According to Reeves, for cylindrical arrays,

> "the output ... is nearly proportional to the amount of solar energy intercepted, and the projected area of a cylinder is $1/\pi$ times the total area. Thus, the cylindrical array should have approximately π times as many cells as a planar array with the same power rating" (Reeves, 1992:317).

Since Modsat requires a great deal of power for some of its applications, the team decided to use planar arrays as opposed to cylindrical arrays. The arrays must be oriented toward the Sun for maximum sunlight incidence. This can be achieved by fixing the panels to the spacecraft body and orienting the body so as to track the Sun with the panels. However, there may not be enough available area on the body of the spacecraft for body-mounted arrays to provide the required power, and thus additional panels must extend from the body. Extended arrays on deployable booms avoid this problem, although they add appendages to the spacecraft which add to the mechanical complexity and take up precious space within the launch vehicle fairing. Moreover, it is easier to place the panels away from payload instruments and other temperature sensitive subsystems which could be damaged from the highly variable temperature environment of solar cells.

In addition, the body-mounted approach uses up two of the three spacecraft axes of control (see section 15.1.3). Recall that optical mission modules must be nadir-pointed, which also requires two axes of control. An axis of control can be freed up by mounting the planar arrays on articulated booms. These booms must work in conjunction with one of the spacecraft's control axes to track and point the Sun for maximum sunlight incidence.

The actual mission modules for each Modsat bus are not known at this stage. Power requirements can vary greatly from module to module, and it is unknown where the sensitive instruments will be. In order to ensure adequate power for all intended missions, accommodate a wide range of mission module design and spacecraft subsystem arrangements, and facilitate the control of nadir-pointed mission equipment, the team decided to use articulated solar arrays on deployable booms.

As mentioned earlier, a power system that accommodates the highest-demand mission modules would be oversized for those mission modules that do not require as much power. In some cases, the arrays would be grossly oversized. It is desirable to minimize the amount of excess power on Modsat. Moreover, it is more economical to limit the size of the solar array structure to the size necessary to produce the required power. Therefore, the team recommends the modular solar array approach used by NASA's Small Explorer (SMEX) Program (Everett, 1996). Modular solar array substrates are also being designed by Composite Optics for possible use on Phillips Laboratory's MightySat II spacecraft (Hagan, 1996).

In the modular approach, a solar array is constructed of smaller solar modules. These modules are essentially small solar panels (8" by 16" for the SMEX program), which can be purchased in mass quantities (at reduced prices) long before assembly. Once the required power for the spacecraft is determined, the appropriate number of modules can be pulled from storage and integrated into a customized solar array assembly. This assembly consists mainly of a structural grid for inserting the solar modules. It must be built so as to conform to the chosen stowed/deployed configurations for the spacecraft. Modular arrays tend to have a slightly lower power conversion efficiency compared to traditional arrays, mainly due to their method of construction and integration within the grid. But this inefficiency is made up for by the reduced cost of solar array expenditures and engineering (Everett, 1996).

The modular approach has the disadvantage of not being tailored to the exact power needs of the spacecraft, since an integer number of solar modules must be used. However, a major purpose of the Modsat concept is to depart from the customized approach in a way that allows flexibility in the design and integration of each spacecraft. Modular solar arrays are ideal for this purpose. Moreover, the traditional approach to designing and building the solar arrays typically requires much advance planning. With the modular approach, the engineers can build the solar array assemblies much later in the overall process of spacecraft construction (Everett, 1996).

Although the modular solar array approach is recommended, the team decided to design each of the alternative Modsat concepts with maximum power for the sake of designing to the most difficult set of requirements.

In section 15.5.3.5, the scheme for stowing the solar arrays during launch was discussed. The arrays will be wrapped around the bus in the stowed configuration. Thus, the amount of surface area provided by the bus, combined with the number of wraps, determines the power generation capability of the arrays, depending on the efficiency of the solar cells.

A study was conducted on three major types of cells: silicon (Si), gallium arsenide (GaAs), and indium phosphide. Performance factors are shown in Table 15-34:

Table 15-34: Single-Cell Performance Comparison for Photovoltaic Solar Cells

Cell Type	**Silicon**	**Gallium Arsenide**	**Indium Phosphide**
Planar cell theoretical efficiency	18%	23%	22%
Achieved efficiency	14%	18%	19%
Equivalent time in geosynchronous orbit for 15% degradation			
- 1 MeV electrons	10 yr	33 yr	155 yr
- 10 MeV protons	2 yr	6 yr	89 yr

Source: McDermott, 1992:397

The design of solar arrays involves tradeoffs between mass, area, cost, and risk. Gallium arsenide and indium phosphide provide better efficiency and longer use than silicon, although they are up to seven times more expensive in terms of cost per watt. Indium phosphide is a relatively immature technology. Given the desire to use current, off-the-shelf technology, minimize cost, and provide a design lifetime of a year, indium phosphide was ruled out. Silicon is a clear choice for consideration. It is a proven, mature technology that has been the industry standard for years, although it is more bulky and less survivable than other materials. Gallium arsenide is also a proven technology, and is becoming more and more common on spacecraft. Although it is expensive compared to silicon, alternatives for which mass and volume become critical may require the use of gallium arsenide.

In the EPS design portion of the Modsat model (see Volume III), one can choose either silicon or gallium arsenide as the solar cell material. However, throughout the design process, it became clear that silicon cells would not provide enough power per area to satisfy the requirement of the CDM (up to 500 watts), given the stowed configuration limitation on the size of the arrays. Thus, all of the Modsat alternatives were designed with gallium arsenide solar cells.

15.5.6.4.3 Design Process

The following process was used to design and size the solar arrays for Modsat.

1. Determine the following factors:
    a. Orbital period
    b. Maximum eclipse duration
    c. Design lifetime
2. Determine the available solar array area.
3. Select the type of solar cell and estimate the power output, with the Sun normal to the surface of the cells.
4. Determine the beginning-of-life (BOL) power production capability per unit area of the array.
5. Determine the end-of-life (EOL) power production capability per unit area of the array.
6. Calculate the amount of raw power that can be produced by the arrays.
7. Calculate the amount of useful power that can be produced by the arrays.
8. Estimate the mass of the solar arrays.

Step 1: The orbital period, maximum eclipse duration, and the period of the complementary daylight portion of the orbit must be calculated for the baseline orbit. This is performed by the Modsat model (see Volume III). The design lifetime of the spacecraft must be chosen. For the purposes of EPS design, a lifetime of one year was chosen to be consistent with the requirements of the CDM. The values for maximum eclipse duration and design lifetime are used later in solar array calculations.

Step 2: The available solar array area, A_{sa}, is determined through structural modeling.

Step 3: Once the type of solar cell has been chosen, the energy-conversion efficiency of the cells is known. From this, and from the illumination intensity of sunlight at normal incidence (1358 W/m^2), the power output capability can be calculated. This power output represents the ideal capability of the cells, and does not take into account losses and degradation due to manufacturing and assembly, known as inherent degradation.

The achieved energy-conversion efficiencies for silicon and gallium arsenide are 14% and 18%, respectively. Thus, the power output, P_0, can be calculated as follows (McDermott, 1992:395):

$$\textbf{Si: } P_0 = 0.14 \times 1358 \text{ W/m}^2 = 190 \text{ W/m}^2 \qquad \textbf{(Eqn 15-26)}$$

$$\textbf{GaAs: } P_0 = 0.18 \times 1358 \text{ W/m}^2 = 244 \text{ W/m}^2 \qquad \textbf{(Eqn 15-27)}$$

Step 4: The BOL power production capability per unit area of the array must be determined from multiplying the ideal power output by factors for inherent degradation and non-normal sunlight incidence. Inherent degradation is contributed to by three basic elements, as shown in Table 15-35.

Table 15-35: Inherent Degradation

Elements of Inherent Degradation	Nominal	Range
Design and assembly	0.85	0.77-0.90
Temperature of array	0.85	0.80-0.98
Shadowing of cells	1.00	0.80-1.00
Inherent Degradation	0.77	0.49-0.88

Source: McDermott, 1992:397

The factor for non-normal sunlight incidence is known as known as cosine loss (McDermott, 1992:400). This can be determined by taking the cosine of the worst-case sun incidence angle, defined between the vector normal to the surface of the array and the Sun line.

BOL power production per unit area, P_{BOL}, is (McDermott, 1992:400):

$$P_{BOL} = P_0 \, I_d \cos\theta \qquad \textbf{(Eqn 15-28)}$$

where I_d is the inherent degradation, and $\cos\theta$ is the cosine loss. For Modsat, the nominal value of I_d, and the worst-case sun angle of $\theta = 23.5$ deg (McDermott, 1992:400) were used.

Step 5: EOL power production capability is determined by multiplying BOL power production per unit area by a factor for lifetime degradation. This degradation is due mainly to radiation damage from the space environment. Such radiation varies depending on the orbital altitude. Other factors contributing to lifetime degradation are thermal cycling, micrometeoroid strikes, plume impingement from thrusters, and material outgassing (McDermott, 1992:400).

In general, for a silicon solar array in LEO, degradation is up to 3.75% per year, while for gallium arsenide it is up to 2.75% per year (McDermott, 1992:400). Cumulative degradation depends on the spacecraft design life.

Lifetime degradation, L_d, can be estimated using (McDermott, 1992:400):

$$L_d = (1 - D)^{SL} \qquad \textbf{(Eqn 15-29)}$$

where D is the degradation per year, and SL is the spacecraft design life. From this, power production per unit area at EOL, P_{EOL}, is (McDermott, 1992:401):

$$P_{EOL} = P_{BOL} L_d \qquad \textbf{(Eqn 15-30)}$$

Step 6: The raw power output of the solar arrays at EOL, P_{raw}, is determined from (McDermott, 1992:401):

$$P_{raw} = A_{sa} P_{EOL} \qquad \textbf{(Eqn 15-31)}$$

Step 7: Not all of the raw power is available, as losses occur in the transmission through wires, batteries, regulators, and converters. Transmission efficiencies depend on the type of power regulation scheme employed by the spacecraft. The two basic types of schemes, discussed in section 15.5.6.7, are direct energy transfer (DET) and peak-power tracking (PPT). DET transmission efficiencies are about 5% to 7% greater than PPT transmission efficiencies because the PPT requires a power converter between the arrays and the loads (McDermott, 1992:396).

The amount of useful power available from the solar arrays at EOL, P, can be estimated from (McDermott, 1992:396)

$$P = \frac{P_{raw} T_d}{\frac{T_e}{X_e} + \frac{T_d}{X_d}} \qquad \textbf{(Eqn 15-32)}$$

where P_e and P_d are the spacecraft's average power requirements (excluding regulation and battery charging losses) during eclipse and daylight, respectively. P_e and P_d were taken to be equal for this study. T_e is the maximum eclipse period, and T_d is the period of the complementary daylight portion of the orbit. The terms X_e and X_d represent the transmission efficiency of the paths from the solar arrays through the batteries to the individual loads and the path directly from the arrays to the loads, respectively. For DET, the efficiencies can be estimated at $X_e = 0.65$ and $X_d = 0.85$; for PPT they are $X_e = 0.60$ and $X_d = 0.80$ (McDermott, 1992:396).

Step 8: The mass of the solar arrays is found by dividing the raw power output of the arrays by their specific performance. The specific performance range of current designs is 14 to 47 W/kg at EOL (Reeves, 1992:317).

A conservative value of specific performance, 25 W/kg, was chosen to calculate the mass of the solar array, M_{sa}. Thus,

$$M_{sa} = 0.04\, P_{sa} \qquad \textbf{(Eqn 15-33)}$$

Example:

1. Let $T_e = 35$ minutes; $T_d = 53$ minutes; SL = 1 year.
2. Let $A_{sa} = 7\ m_2$
3. Choose gallium arsenide solar cells, so that $P_0 = 244$ watts/m_2; $D = 0.0275$
4. Let $I_d = 0.77$; $\theta = 23.5$ deg; thus, $P_{BOL} = 172$ watts/m_2
5. $L_d = 0.9725$; $P_{EOL} = 167$ watts/m_2
6. $P_{raw} = 1169$ watts
7. Design for a DET system; thus, $X_e = 0.65$; $X_d = 0.85$; $\Rightarrow P = 533$ watts

A final consideration for the solar arrays is the selection of solar array drive motors (SADMs). SADMs are necessary to mechanically rotate the solar array structures to achieve optimum angles of incidence with sunlight. Two SADMs are needed, one for each solar array. For modeling purposes, representative SADMs were chosen from the Jet Propulsion Laboratory (JPL) Flight Hardware Survey, with the following characteristics:

- Mass: 5 kg each.
- Dimensions: Each is a cylinder, 10 cm diameter by 27 cm length.

15.5.6.5 Energy Storage

15.5.6.5.1 System Synthesis

Since Modsat will not be generating power with a constant source, such as radioisotopes or nuclear reaction, a system must be designed to store energy for eclipse operations. The storage system must also be able to handle peak loads beyond the capability of the solar arrays, which are designed to handle average loads.

The most common storage devices for LEO spacecraft are chemical batteries, known as secondary batteries since they discharge during eclipse and recharge in sunlight (McDermott, 1992:402). Other storage schemes are available, such as thermal storage, but they are far less common for LEO spacecraft than batteries. Some new approaches, such as flywheel storage, require much less weight than batteries. However, it was decided that new approaches with unproven technology would not be appropriate for the Modsat EPS. Space batteries are commonly used and widely available, and are therefore appropriate for Modsat energy storage.

15.5.6.5.2 Tradeoffs and Decisions

The two primary types of batteries in use today are nickel cadmium (NiCd) and nickel hydrogen (NiH_2) batteries. Specific energy density ranges for these battery types are shown in Table 15-36.

Table 15-36: Specific Energy Densities of Selected Battery Types

Battery Type	Specific Energy Density (W hr/kg)
Nickel cadmium	25-30
Nickel hydrogen (individual pressure vessel design)	25-40
Nickel hydrogen (common pressure vessel design)	45-60

Source: McDermott, 1992:403

Nickel cadmium is a proven, easily available and comparatively cheap technology, but it delivers a relatively low energy density. Nickel hydrogen delivers more energy per kilogram than nickel cadmium, but it is more expensive. Moreover, it is a less mature technology for LEO applications (McDermott, 1992:403). However, in recent years nickel hydrogen has been used successfully in many spacecraft. The new common pressure vessel (CPV) design nickel hydrogen technology was successfully qualified on the Clementine mission in 1994 (Clementine Report).

If the minimization of cost was the highest-priority objective in component selection, nickel cadmium batteries would be recommended for Modsat. However, the cost objective received a lower priority than that of mission utility. In addition, since the CDM requested a peak power capability of 1000 watts, Modsat requires a great deal of battery capacity. The weight of the bus must be minimized; therefore, nickel hydrogen was preferred by the team. In fact, in an effort to drive down weight, the team decided to design all alternatives with two nickel hydrogen CPV batteries from Johnson Controls, the same battery flown on the Clementine mission. Characteristics of this battery are shown in Table 15-37. This design choice gives Modsat 30 amp-hours of battery capacity.

Table 15-37: Characteristics of the Johnson Controls/Clementine NiH_2 CPV Battery

# of cells	Capacity (amp-hour)	Energy density (W hr/kg)	Dimensions (cm)	Weight (kg)
22	15	47.1	50.8 x 13.4 x 13.4	9.5

Source: Clementine Report

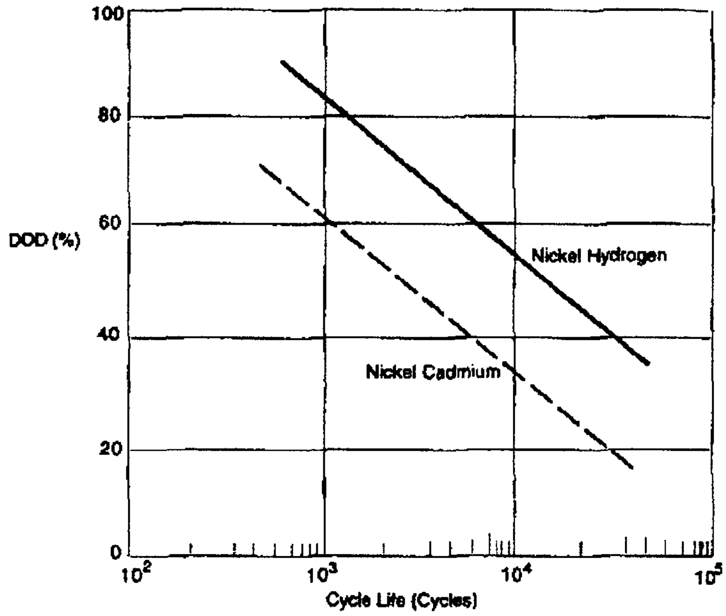
It should be noted that future design efforts on Modsat could result in the selection of different batteries. The Modsat model allows the user to choose from a list of batteries (see Volume III). As currently designed, the Modsat alternatives do not offer modularity in the power storage system. The NiH_2 batteries are large fixtures within the spacecraft, and would be very difficult to remove and replace. More flexible alternatives were examined, such as the use of a number of smaller batteries to provide the same capacity. Such an arrangement would potentially allow for the removal of unneeded batteries for less demanding applications. For instance, the SMEX program is currently designing spacecraft busses with small 4 amp-hour NiCd battery packs (Everett, 1996). SMEX engineers are able to select the number of required batteries for a given mission.

Although such modularity would provide the flexibility sought by the CDM, it has its disadvantages. A modular scheme would likely employ several low-capacity batteries. Research into currently used batteries revealed that low-capacity batteries are mostly available in the NiCd variety; a set of NiCd batteries could weigh twice as much as a NiH_2 CPV configuration, for the same capacity. Moreover, the use of many batteries adds to the complexity of battery charge/discharge mechanics and control. In keeping with the importance of minimizing weight and maximizing capability, the team traded off modularity for a low weight, high power battery choice.

15.5.6.5.3 Design Process

Spacecraft bus voltage is determined by the voltage of the batteries. Typically, spacecraft maintain a 28 volt bus, from which power is converted and further regulated as needed for individual loads. The voltage of a battery is determined by the number of cells in series, while the capacity (watt-hrs or amp-hrs) is determined by the number of cells in parallel (McDermott, 1992:401). Thus, batteries can be connected in series or parallel to provide more voltage and capacity, respectively. A 28-V aerospace battery usually consists of 22-23 series-connected cells (McDermott, 1992:403).

The orbit of the spacecraft determines the number of charge/discharge cycles the batteries must support during the lifetime of the spacecraft. For example, a LEO spacecraft with an orbital period of 90 minutes would orbit the earth 5840 times in one year. Since an eclipse occurs in every orbit at such a low altitude, the batteries would experience 5840 charge/discharge cycles. A characteristic of secondary batteries is that the allowable depth-of-discharge (percentage of capacity which can be used during a cycle) decreases logarithmically with the lifetime number of charge/discharge cycles, as shown in Figure 15-43.



Source: McDermott, 1992:404

Figure 15-43: Depth-of-Discharge vs. Cycle Life for Batteries

Once the number of cycles and average DOD are known, the required battery capacity can be determined.

The EPS engineer must ensure that the batteries can provide the required power during every eclipse. The highest capacity is required during the maximum eclipse period, T_e. This capacity is calculated as P_eT_e, where P_e is the required average eclipse load. The batteries must also be able to supplement the solar arrays in the provision of the peak power requirement. The corresponding required capacity is P_pT_p, where P_p is the difference between the peak load requirement and the average power supplied by the solar arrays, and T_p is the amount of time during which peak loads are required. The batteries should be sized to accommodate the largest of the two capacities discussed above.

The value for T_p must be specified as a requirement. As no such value was specified for this project, the team decided to model Modsat's peak power period requirement after an existing program of a similar nature. Phillips Laboratory's MightySat program is developing a standardized bus for space experiments. The peak power period for MightySat is specified to be 80% of the average daylight period (MightySat TRD). The team adopted this requirement for Modsat, with the exception that the daylight period used is T_d, the complement of the maximum eclipse period. Thus, for Modsat, $T_p = 0.8T_d$.

The capacity requirement for the batteries is the ideal capacity, C_i, that the batteries would have to supply if there were no battery-to-load transmission losses. This must be increased to account for DOD limitations and the battery-to-load transmission efficiency, n. For this study, the batteries were sized based on a transmission efficiency of 90%. If the chosen number of batteries is N, the required capacity per battery is (McDermott, 1992:405)

$$C_r = \frac{C_i}{(DOD)Nn} W - hr \qquad \textbf{(Eqn 15-34)}$$

where C_i is in watt-hours. The corresponding capacity in amp-hrs is determined by dividing the W-hr capacity by the bus voltage, taken to be 28-V for this study.

Example:

1. Let P_e = 500 W; peak load = 1000 W; T_e = 0.6 hr; T_p = 0.7 hr; DOD = 0.55; N = 1
2. P_p = 500 W $\Rightarrow$ C_i = 350 W-hr
3. C_r = 350/(0.55•0.9) = 707 W-hr = 25.25 amp-hr

The mass of the batteries is found by dividing the total storage capacity by the specific energy density (W-hr/kg) of the type of battery.

15.5.6.6 Power Distribution

15.5.6.6.1 System Synthesis

The power distribution system consists of the cabling, fault protection, and power switching gear to turn power on and off to the spacecraft loads. A major focus in distribution system design is on keeping mass and power losses at a minimum while providing survivability, cost, reliability, and power quality.

Distribution systems are highly unique for each spacecraft, since each has its own unique arrangement of subsystems and mission modules. Thus, the load profile of the spacecraft must be known in order to design an efficient distribution system.

However, the Modsat concept presents a unique challenge to the electrical engineer. Since it is intended to be a generic, standardized bus, the required load of the mission equipment is unknown until the mission need arises. Moreover, flexibility in configuration is a key value of the decision maker. Therefore, there is no fixed load profile for Modsat; the distribution system must take this into consideration.

The discussion below focuses on power switching and distribution. Power switches may either be standard mechanical relays, or solid-state relays based on semiconductor technology (McDermott, 1992:405). The power distribution architecture may be centralized or decentralized, depending on the location of the power converters. The centralized approach implies a regulated bus, with a unique suite of converters at a central location that distribute converted, regulated power to known, fixed loads. The decentralized approach implies an unregulated bus, with conversion and regulation occurring at each load user (McDermott, 1992:405) .

15.5.6.6.2 Tradeoffs and Decisions

Mechanical relays are the clear choice "because of their proven flight history, reliability, and low power dissipation" (McDermott, 1992:405). Solid-state relays may be the standard choice in the future, but presently they are an immature space technology. Since minimizing technological risk is important to the decision maker, solid-state relays were ruled out.

Decentralized distribution is the best approach for Modsat. It allows for a great deal of flexibility, thereby complementing a modular architecture. Each power-using component, including the mission module, must provide its own converter/regulator. The load nodes will be fixed, but the load users may be changed. This scheme obviates the need for customizing the distribution network for each mission application.

Another consideration for power distribution is battery reconditioning. Battery reconditioning is a process which completely drains a battery in order to eliminate loss of capacity due to continuous cycling at a limited depth-of-discharge. It requires a battery to be taken completely off-line, drained, and recharged. Battery reconditioning is an option for the designer which, if chosen, requires extra relays and shunt resistors to bleed the batteries and dissipate their energy. Due to the short required lifetime of Modsat, battery reconditioning has not been considered.

15.5.6.6.3 Design Process

The distribution system should include provisions for fault detection, isolation, and correction during testing (McDermott, 1992:406). Each fully integrated Modsat spacecraft must undergo extensive testing, especially since each many configurations will be unique. A full set of fuses, placed in series with the power bus, will be required for fault testing (McDermott, 1992:407).

The mass of the distribution system, excluding wiring, can be estimated at 0.02 kg per watt of distributed power (Reeves, 1992:319). The wiring can have up to 4% of the mass of the dry spacecraft (Reeves, 1992:319). Since this is not a trivial amount, it is important to keep the distribution cables as short as possible.

For modeling purposes, a power control unit (PCU) was chosen from JPL's Flight Hardware Survey to represent the distribution system. This unit is a 24 x 16 x 20 cm box.

15.5.6.7 Power Regulation and Control

15.5.6.7.1 System Synthesis

Power regulation must be considered for three key elements: controlling the solar array, regulating bus voltage, and charging the battery (McDermott, 1992:407). Solar array power must be controlled at the array to prevent battery overcharging and excessive spacecraft heating (McDermott, 1992:407). Two primary schemes are available: a peak-power tracker (PPT) and a direct-energy-transfer (DET) system (McDermott, 1992:407; Everett, 1996). A PPT is nondissipative; it extracts the exact amount of required power up to the array's peak output. A PPT operates in series with the solar array, and uses 4-7% of the total power (McDermott, 1992:407). The DET system is dissipative because it shunts all power not used by the loads. A shunt regulator operates in parallel to the array and shunts the array current away when the loads or battery charging do not require the power (McDermott, 1992:407).

Bus voltage may be unregulated, quasi-regulated, or fully regulated (McDermott, 1992:407). In an unregulated scheme, bus voltage variation equals battery voltage variation, which can be as high as 20% between charge and discharge (McDermott, 1992:407). A quasi-regulated system regulates bus voltage during battery charge but not during battery discharge. A fully regulated system employs both charge and discharge regulators.

Batteries can be charged individually or in parallel. Parallel charging keeps the voltage of all batteries the same, but allows current and temperature to vary (McDermott, 1992:409). Individual charging "optimizes the battery use by charging all the batteries to their own unique limits" (McDermott, 1992:409).

15.5.6.7.2 Tradeoffs and Decisions

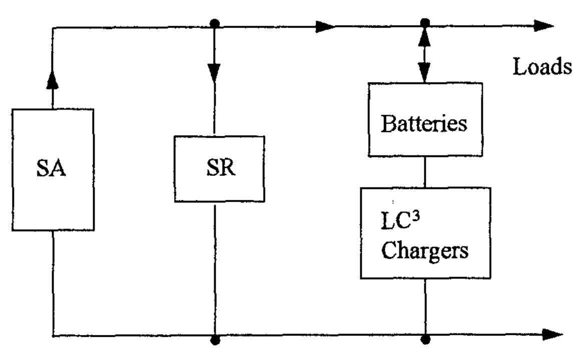
A DET system should be used as opposed to a PPT. The DET has fewer parts, lower mass, and higher total efficiency (McDermott, 1992:407). These qualities are highly desirable within the Modsat value system. It is unknown exactly how much power will be required for each configuration of Modsat. Thus, it seems unlikely that one PPT could be designed for all applications. Moreover, given the use of modular solar arrays, different Modsat missions will have different amounts of power being supplied. A simple shunt-regulated system, i.e., the DET system, would aid in the integration of such an architecture.

For reasons discussed in section 15.5.6.6, an unregulated system should be used. Among the advantages discussed above is the fact that more regulation involves dissipation of energy. This is inefficient and could potentially create heat in undesirable places.

Due to the emphasis on design flexibility, it seems advantageous to employ independent battery chargers. Their use aids in vehicle integration and maximizes the use of each battery (McDermott, 1992:409). Future iterations of the Modsat design may adopt a modular power storage approach, in a scheme where a battery could be removed if not needed. This approach is greatly facilitated by the use of independent charging, and greatly hampered by parallel charging (Everett, 1996). On the other hand, independent charging is more expensive and complicated than parallel charging (McDermott, 1992:409), and adds more weight. Although the choice between parallel and independent charging may not be an issue for Modsat concept exploration, the current tradeoff of cost for mission utility led the team to recommend the independent charging approach.

15.5.6.7.3 Design Process

An example of an unregulated bus using a shunt regulator and an independent charging scheme known as linear, charge-current-control (LC^3), is diagrammed in Figure 15-44. (McDermott, 1992:409). Note: SA = solar array; SR = shunt regulator.



Source: McDermott, 1992:408

Figure 15-44: Shunt Regulator

The mass of the regulator equipment can be estimated at 0.025 kg per watt of regulated power (Reeves, 1992:319).

For modeling purposes, a shunt regulator was chosen from JPL's Flight Hardware Survey to represent the regulation system. This unit is a 42 x 18 x 11 cm box.

15.5.6.8 Future Effort

The Modsat EPS must be integrated into the whole Modsat bus in an intelligent manner. Further design effort for Modsat must take these elements into consideration:

- Electrical loads must be supplied to the mission module and to the other components in the bus.
- EPS must have a command and telemetry interface. Designers must eventually determine which EPS components require commanding, and which must report health and status. An appropriate collection node must be integrated into the command and data handling architecture of Modsat.
- Each component has an operating temperature range. All components must be physically arranged and thermally controlled to maintain an adequate thermal environment for the spacecraft. Some typical operating ranges are listed in Table 15-38.
- EPS components radiate heat, as energy is dissipated through transmission, regulation, etc. The components must be located so that this heat does not negatively affect other sensitive spacecraft components. For instance, the shunt regulator for the solar arrays could dissipate large amounts of unneeded energy. It is necessary to radiate this heat directly into space, away from the spacecraft, wherever possible.

- The spacecraft will have various power switching requirements, both routine and for redundancy purposes. The EPS must incorporate the appropriate equipment to handle this function.
- The spacecraft operating system must manage EPS operations. For example, there must be a way to control the use of the batteries when they are needed. Another example is the process of load shedding during an emergency, whereby power is turned off to selected subsystems in an effort to conserve power.

Table 15-38: Operating Temperature Range

Components	*Typical Temperature Range, °C*
Electronics	0 to 40
Batteries	5 to 20
Solar Arrays	-100 to +100

Source: McMordie, 1995:410

As various spacecraft power technologies become qualified and proven over the next few years, their use within Modsat should be examined. It is likely that industry will develop some attractive alternatives to the traditional menu (i.e., light-weight flywheels for energy storage). Such technologies may be well suited to the standardized bus approach.

15.6 Tradeoff Summary

15.6.1 System Level

- One satellite per launch vehicle
- 3-axis stabilization
- Modular component interfaces
- On-board propulsion

- Mission data storage
- Spacecraft to Payload Interface Guideline (SPIG) interface

15.6.2 Subsystem Level

ATTITUDE DETERMINATION AND CONTROL

- Inertial Measurement Unit
- Star sensor
- Earth sensor
- Reaction wheels

PROPULSION

- Monopropellant configuration
- Hydrazine blend propellant; 24% HN, 76% N_2H_4
- Blowdown pressure system
- Positive expulsion fuel storage system
- Bottom level location for the system
- cylindrical tanks
- thrusters

STRUCTURES

- Octagon "cage" structures with "polysat" design
- Modular mounting plates

THERMAL

- Interface thermal blanket between the mission module and satellite bus

TTC/CDH

- Satellite Operating System

  >> software handles interrupt requests to collect telemetry and handle housekeeping

- Internet communication format (TCP/IP)

  >> JAVA compatible architecture enables user in field to access data via browser capable system

  >> provides point-and-click environment to control satellite

  >> encryption on demand with software plug-in modules

EPS

- Modular solar arrays on deployable, articulated booms
- Solar arrays wrap around the bus in the stowed launch configuration
- Gallium arsenide solar cells
- NiH_2 common pressure vessel batteries
- Decentralized, unregulated distribution
- Shunt regulation; individual battery charging

16. Modeling

Modeling plays an important role in solving complex and multiple variable problems; as such, it is a critical element of systems engineering. Upon using systems engineering to design a standardized tactical satellite bus, the team was faced with many variables and the relationships between them. Satellite design by its very nature is quite complicated, and trying to account for every detail in a preliminary study is infeasible. Modeling enabled the team to approach the design at a high level, focusing only on the major elements of the spacecraft. The team also used the model to evaluate the performance of the alternative solutions.

Before considering modeling methods, it was important to focus on the problem definition, objectives, and measures of effectiveness (MOEs). This ensures that the model is fully relevant to the problem. The model must be able to evaluate the alternatives proposed to solve the problem, within the framework established for that evaluation.

Once the basis for the Modsat model was understood, the team outlined the scope and type of model necessary to meet the objectives. Starting from a high level, the team determined that the Modsat model must be highly integrated (able to perform both design and evaluation) and adaptable. The various elements of the model (i.e., subsystem design modules) must be compatible with the overall model and with each other. The "integrated and compatible" requirement ensures that all of the subsystem designs created in the model can be brought together and combined into a proposed satellite design for further analysis. "Adaptability" means the satellite designer can easily modify and/or expand the model to change its analysis and design characteristics.

In searching for an integrated model, the team queried the Internet and investigated software maintained by AFIT, without success. Therefore, to fulfill the integrated model requirement, the team turned to Matlab, a mathematical and graphics software package, to develop an in-house satellite design model.

In addition to integrating all the various subsystems, the model must test the integrated satellite designs to ensure they meet launch and on-orbit operational requirements. It must also determine how well the proposed alternative meets the criteria established in the value system.

In creating the model to meet all these requirements, the team started with a high-level structure as shown in Figure 16-1.

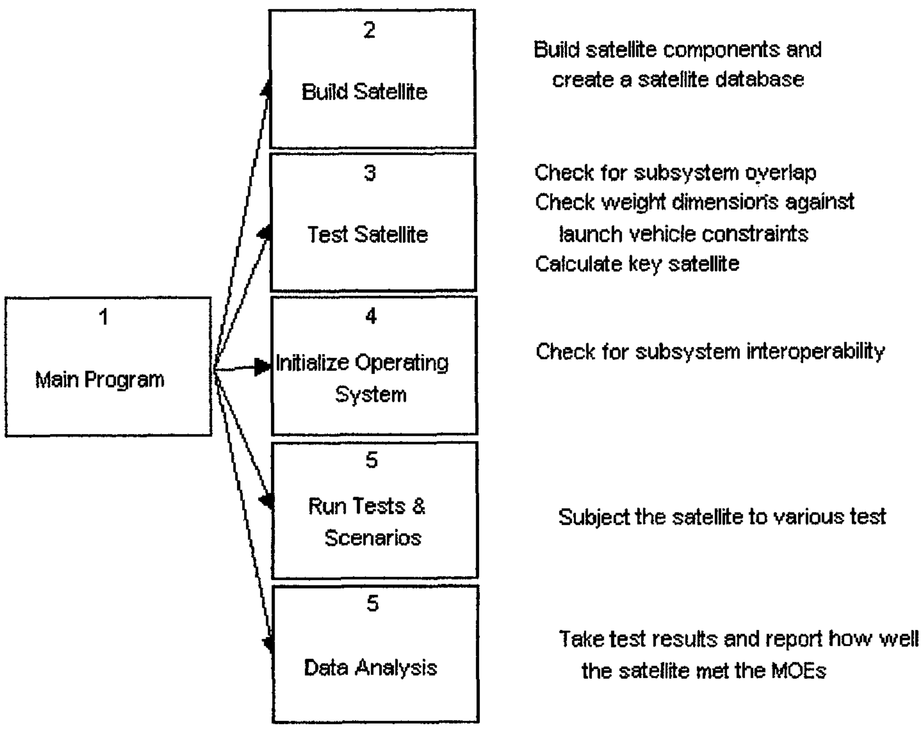


Figure 16-1: Logical Flow of Modsat Model

The Modsat model is geared toward sequential operation, starting from the top of the logic flow diagram and working down. Building satellite components and bringing them together in one integrated satellite database is the first step. Next, Modsat checks the satellite database to ensure total satellite mass, center of mass, and sizing meet the launch vehicle constraints. Once the satellite database passes these physical constraints, it must also pass operating requirements. In this same step cost and reliability are determined. At this point the satellite is integrated, and the subcomponents will physically fit together within the launch vehicle, but it is not known how well the satellite will perform in the space environment. Therefore, the satellite bus must be subjected to launch and orbit tests to determine how well this particular satellite configuration will do. These results are fed into data analysis to be evaluated against the objectives. Each design receives an overall utility score, and the designs are ranked from highest to lowest. Finally, sensitivity analysis is performed by varying the top level objective weights. This shows how well a design will do when subjected to changing environments.

Although the Modsat model uses first order assumptions and calculations, it proved to be an excellent systems engineering tool. For a full description of the model, see Volume III.

17. System Synthesis

The first iteration of System Synthesis concentrated on large-scale architectures as opposed to individual satellite designs. These architectures philosophically captured the different approaches to constellation employment available to the designer of a standard satellite concept capable of supporting multiple tactical users and their requirements. Further satellite design research refocused the system design effort to a narrower, more detailed scope. This focused approach concentrated on the goal of generating the best design for a single, standard spacecraft bus. As the large-scale architectural concepts do not preclude the use of standardized components amongst them, the "baseline" or "one size fits all" architecture became the focus of design efforts.

The original architectural themes presented early in the system design process lend themselves to serve as implementation schemes applicable to the standard design (i.e., "evolving" the baseline bus design). With a standard design in hand, application of architectural themes to this design may further enhance that design's ability to support specific mission modules (if not also improve its overall performance).

17.1 Subsystem Baselines

Taken as a whole, the design options available for the design of spacecraft buses are myriad. The classical morphological system synthesis methodology is too cumbersome to be used in this case, given the design objectives of bus standardization, tactical responsiveness, and adaptable application (not to mention the time frame in which the system process was applied). The design philosophy adopted for the generation of candidate spacecraft bus designs begins with the tradeoff studies presented by each of the designated subsystem experts. These tradeoff studies at each subsystem level served to narrow the candidate design solution space, focusing on design points which, when applied to different designs, would exhibit the greatest variances in utility or represent the greatest differences in approaches to certain bus capabilities.

Each subsystem level study addressed the selection of individual subcomponents and the narrowing of requirements for each satellite subsystem. Most of these choices then became fixed in all of the candidate designs. Each design would then yield a baseline design from which alternative configurations could be based, by varying the characteristics and/or configurations of the subsystems not fully determined by the trade studies. This streamlined approach to subsystem and eventual system design facilitated the focusing of the effort and the effective utilization of extensive satellite subsystem expertise by the more experienced team members.

17.2 Alternative Design Descriptions

The following discussion details the alternative designs evaluated using the Modsat computer modeling, design, and analysis software. To begin scoping the alternatives, all were designed to be compatible with the Pegasus XL/without HAPS configuration. Of the seven designs, six are based on the 38" interface, while the seventh is based on the 23" interface. The team judged early in the design process that the 23" interface configuration probably would not provide enough volume for Modsat because it reduces volume for the mission module and the size of the solar arrays. However, one alternative with this interface was generated for the sake of a complete analysis.

The designs fall into one of two categories. In the first category, there are three levels for component placement. The third level has a Supplemental Mission Adaptive Shelf, or "SMASH", which is a large volume of space reserved for either a dedicated high data rate antenna and its transponder, or some other user-specified payload or mission-unique equipment. The second category has only two levels, reducing the overall bus height by not including a SMASH for additional mission equipment (such as a high data rate communications package). In the case of the second category designs, all mission-unique equipment would be placed entirely on the payload interface plate (very top of the bus).

All designs fit a particular tactical profile, with tactical capability being determined by the amount of fuel on-board. Those designs with more fuel have more ability to make in plane or out of plane maneuvers, satellite altitude corrections, or other maneuvers. In-plane changes can be measured by ΔV; thus, total ΔV capability is the measure for how much a satellite can change its velocity. ΔV capability is directly proportional to the amount of fuel carried by the spacecraft. All of the designs have varying amounts of ΔV capability according to three sizes of propellant tanks carried by the spacecraft. There are three profiles for ΔV: max (450 m/s), mid (300 m/s), and low (100 m/s). These tactical profiles correspond to varying demands which may or may not be placed on the satellite during its mission. For a given category the three tactical profiles are essentially the same except for the size of the propellant tanks. Since the bottom level of the bus is reserved for propulsion, different tank sizes will raise or lower the bottom mounting plate, thus increasing/decreasing the overall height of the satellite bus. The three tactical profiles and their capabilities are summarized in Table 17-39:

Table 17-39: Tactical profiles

MAX	Maximum ΔV capacity for orbit maintenance Up to 2.5 degrees of inclination change during mission
MID	Medium ΔV capacity for orbit maintenance Up to 1.5 degrees of inclination change during mission
LOW	Minimum ΔV capacity for orbit maintenance (one year)

All designs have the same power supply capability (447 Watts average/1000 Watts peak), except for the 23" interface design, which has lower output (390 Watts avg, 950 Watts peak) due to reduced solar array area. The seven designs generated as alternative solutions are as follows:

- MAXTAC: Max tactical profile, with three component levels and SMASH
- MIDTAC: Mid tactical profile, with three component levels and SMASH
- LOWTAC: Low tactical profile, with three component levels and SMASH
- MAXTAC-N: Max tactical profile, with two component levels and no SMASH
- MIDTAC-N: Mid tactical profile, with two component levels and no SMASH
- LOWTAC-N: Low tactical profile, with two component levels and no SMASH
- MIDTAC-23: Mid tactical profile, with three component levels and SMASH; 23 inch interface

The major characteristics of each design, including the height of each bus, were determined through modeling, and the results are shown in Table 17-40.

Table 17-40: Major Design Characteristics

Acronym	SMASH	Number of Levels	ΔV (m/s)	Interface (in)	Bus Height (cm)
MAXTAC	Yes	3	450	38"	71
MIDTAC	Yes	3	300	38"	68
LOWTAC	Yes	3	100	38"	63
MAXTAC-N	No	2	450	38"	69
MIDTAC-N	No	2	300	38"	66
LOWTAC-N	No	2	100	38"	61
MIDTAC-23	Yes	3	300	23"	95.7

17.3 Convergence of Individual Designs

The system trades, subsystem trades, and the category/tactical profile options together form the basis for a set of feasible, alternative satellite designs. Based on the research done at the subsystem level, the team was able to develop a high-level set of required components and characteristics for Modsat (see section 17.4). However, an actual satellite design is an integrated system, and not simply a collection of components individually meeting requirements. The overall integrated satellite design and concepts must meet the requirements for this design study. In order to effectively evaluate the alternatives, they must each form a complete satellite design, wherein components are properly positioned within the structure of the satellite.

Although the Modsat designs are very similar, they differ by category, tactical profile, and launch configuration. The team developed a "baseline" Modsat design by determining the optimal placement of subcomponents. From this "baseline" design, other designs were created by simply varying the satellite configuration to accommodate the various category (SMASH/no SMASH) and tactical profile (propulsion) options. The design most divergent from the rest, of course, is the design based upon the Pegasus 23-inch interface.

The process of satellite design convergence began with the Pegasus payload bay size, weight, and center of gravity constraints. While the structure was being developed to fit within the payload bay, the major design effort focused on the problem of intelligently fitting and integrating the subcomponents within it while maximizing the remaining volume for the mission module. The main size constraint is the payload bay's diameter and overall height, since the mission module/satellite bus must physically fit within the LV fairing shown below in Figure 17-1.

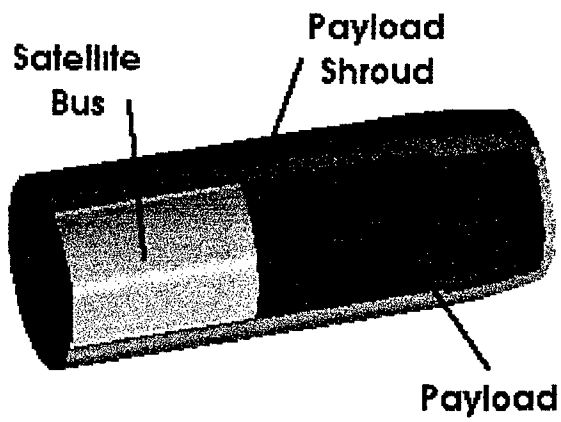


Figure 17-1: Typical Launch Vehicle Shroud Configuration

Starting with the interface plate of the launch vehicle and proceeding upwards with successive component "mounting plates," the placement of components was initially estimated, spatially evaluated, and iteratively adjusted, until a reasonable, desirable arrangement of components solidified into an integrated satellite design. The semi-cylindrical, shelf-like structure chosen for Modsat is conducive to this design convergence approach and facilitates the organization of components into "functional" groups.

It should be noted that the particular component placement schemes chosen by the team are not intended to be set in stone. The team used common sense to arrange the equipment within the bus, though no member of the team had any prior experience designing spacecraft. Future iterations on the design of Modsat may very well yield altered component arrangements. Moreover, only those major components analyzed in this study were placed in the bus. A more detailed design effort must include a more complete set of spacecraft hardware.

17.4 Component/Characteristic Listing

The following list summarizes the set of characteristics and components that were chosen for Modsat. It should be clear that, to be consistent with the scope of this study, this is a high-level set of major components. The rationale behind the selection of these components is discussed in the Tradeoffs section of this report. Only those characteristics that were relevant to the integration of the overall design are included below.

- Launch Vehicle Fairing Static Envelope (inside of which the spacecraft may be placed)
  Note: The Pegasus User's Guide specifies a dynamic envelope of 46". The team decided to use a more conservative static envelope due to the risk inherent in the use of the wrap-around solar array assemblies. This approach is used by the SMEX program (Everett, 1996).

    - Diameter: 44" ≈ 112 cm

    - Height from spacecraft interface to curve in fairing: 111 cm

- Structures and Mechanisms
  Note: All structural members are made of 7075-T6 aluminum, with a density of 2,800 kg/m^3.

    - Shape: Octagon

- Structural cylindrical columns (8)
  - Dimensions: hollow; 4 cm outer diameter; 1.5 cm inner diameter; height varies with each design

- LV interface plate
  - Diameter: 111.76 cm
  - Thickness: 1 cm

- Component mounting plates (2 for with SMASH, 1 for no SMASH; note that the LV interface plate forms the bottom component plate)
  - Diameter: 87.26 cm
  - Thickness: 0.2 cm

- Center spine
  - Dimensions: 5 cm square; hollow; height varies with each design

- Propulsion

  - Propellant tanks (4)
    - Dimensions: hollow cylinder; 2 tanks are 42 cm long, 2 are 33 cm long; diameter varies with tactical profile

  - Thrusters (6)
    - Dimensions: modeled as a cylinder; 6.6 cm diameter; 10.2 cm height

  - Valves (6)
    - Dimensions: modeled as a cylinder; 6.6 cm diameter; 8.2 cm height

- ADCS

  - Reaction wheels (4)
    - Dimensions: cylinder; 25.5 cm diameter; 9 cm height
    - Mass: 5.1 kg each
    - Power requirement: 17 watts max

  - Reaction wheel electronics boxes (4)
    - Dimensions: box; 17.8 by 15.2 by 3.2 cm
    - Mass: 0.9 kg each

  - Earth sensor (head)
    - Dimensions: cylinder; 10.4 cm diameter; 16.3 cm length
    - Mass: 1.27 kg
    - Power requirement: 0.5 watts

- Earth sensor (electronics)
  - Dimensions: box; 10.2 by 20.3 by 6.7 cm
  - Mass: 1.14 kg
  - Power requirement: 3.5 watts

- Star sensor
  - Dimensions: cylinder; 13.5 cm diameter; 14.2 cm length
  - Mass: 2.5 kg
  - Power requirement: 10 watts

- Inertial Measurement Unit (IMU)
  - Dimensions: cylinder; 21.6 cm diameter; 13.3 cm height
  - Mass: 3.72 kg
  - Power requirement: 33 watts

- EPS

  - Solar array panels (16)
    Note: Since the bus has an octagon shape, there are eight hinged panels per wrap around the bus. All configurations used two wraps, therefore all have 16 panels.
    - Dimensions: box
      -- Thickness: 4 cm
      -- Width: inner wrap panels are 36.14 cm wide; outer wrap panels are 39.45 cm wide
      -- Height: varies with each design
    - Mass: varies with size of arrays

  - Batteries (2 NiH_2)
    - Dimensions: cylinder; 13.4 cm diameter; 50.8 cm length
    - Mass: 9.5 each
    - Capacity: 15 amp-hours each

  - Regulator
    - Dimensions: box; 42 by 18 by 11 cm
    - Mass: varies with power output

  - Power Control Unit (PCU)
    - Dimensions: box; 24 by 16 by 20
    - Mass: varies with power output

  - Solar Array Drive Motors (2 SADMs)
    - Dimensions: cylinder; 10 cm diameter; 27 cm length
    - Mass: 5 kg each

- TT&C/CDH

  - Transceiver
    - Dimensions: box; 21 by 15 by 13 cm
    - Mass: 3.41 kg
    - Power requirement: 22 watts

  - SGLS Antennas (2)
    - Dimensions: cylinder; 4 cm diameter; 10.2 cm height
    - Mass: 0.25 kg each

  - Central Processing Unit (CPU)
    - Dimensions: box; 5 by 22 by 15
    - Mass: 0.9 kg
    - Power requirement: 2.8 watts

  - Data Storage Unit
    - Dimensions: box; 13 by 13 by 13
    - Mass: 3.4 kg
    - Power requirement: 1 watt

17.5 Baseline Design

The MAXTAC bus (maximum propulsion, three shelves with SMASH) was chosen as the baseline upon which to conduct the detailed placement of components. The following discussion applies to the design of the MAXTAC. In section 17.6, the other alternatives will be discussed. The complete set of databases for each design is included with the modeling software (see Volume III).

With the basic structure at hand, the team had to choose desired locations for the components. It was decided to place the components as shown in Table 17-41.

Table 17-41: Component Placement

Component Level	Components
1st	Propulsion; one SGLS antenna
2nd	Batteries; PCU; Regulator; Reaction wheels and electronics; Earth sensor and electronics; Star sensor
3rd	IMU; TTC & CPU components; SADMs

The rationale for locating the propulsion subsystem on the bottom level is documented in the Tradeoffs section of this report. It was decided to place a SGLS antenna on the bottom level as well, such that the antenna would have a field of view through a hole in the LV interface plate. It was felt that an antenna was necessary in this location in order to communicate with Modsat prior to its full structural deployment.

The team decided to place components on the second level in such a way as to keep the spacecraft center of mass as low as possible (due to LV considerations). Thus, the relatively massive batteries and reaction wheels were placed on the second level. In order to collocate as many subsystem components as possible, the regulator, CPU, reaction wheel electronics, earth sensor and electronics, and star sensor were placed on the second level as well.

However, it was necessary to place the SADMs on the third level. The axis of the SADMs must be aligned with the axis of the solar array assemblies, which must in turn be located high enough to provide a balanced inertia matrix for the total spacecraft. In addition, the SADMs must be placed on the outer edge of the component plate, in order to mate with the extended solar array booms. The remaining TT&C/CPU components were placed on the third level. Nearly one half of the third level was reserved for the

SMASH. The team modeled the use of this space with a hypothetical high data rate (HDR) package, consisting of an antenna and a transceiver. The antenna was modeled as a large cylinder, while the transceiver was chosen to be identical to the TT&C transceiver.

The very top mounting surface of the bus (designated the "payload interface plate") serves as the substructure for the mission modules. Several different types of mission modules may be evaluated with the various bus designs by using the Modsat computer model. For ease of evaluation for this design study, the mission modules are oriented in the axial direction because the on-orbit attitude for the Modsat will have the payload interface plate pointing nadir. Future design efforts may introduce alternative orientations.

Finally, it was necessary to keep the spacecraft center of mass as close as possible to its central axis. Thus, mass symmetry was a major factor in the placement of components.

17.5.1 Structure

Based upon the ΔV requirements given for the tactical profile desired, the height of the first component plate was set, and the initial positioning of components could commence. The MAXTAC structure is shown in Figure 17-2.

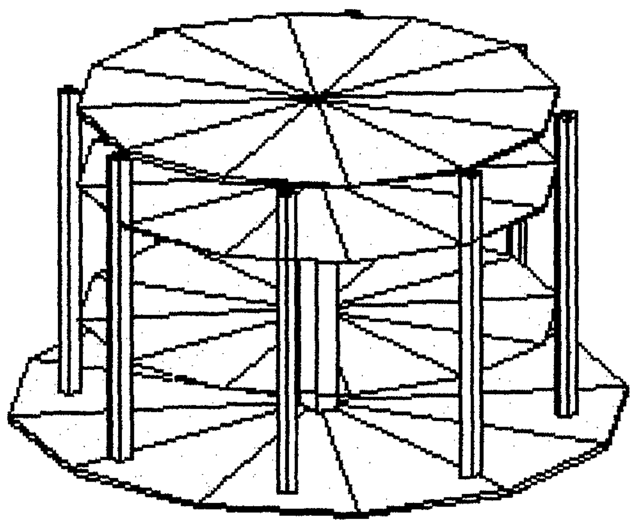


Figure 17-2: MAXTAC Structure

17.5.2 Propulsion

Figure 17-3 shows the propulsion subsystem integrated into the structure.

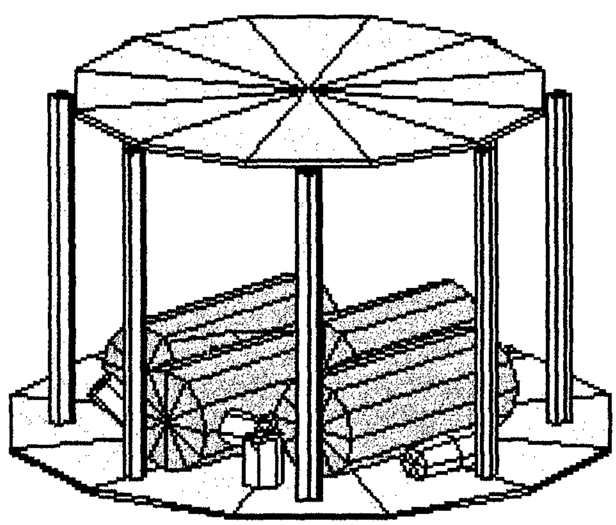


Figure 17-3: MAXTAC Propulsion Subsystem

17.5.3 ADCS

For the purpose of optimum three-axis control, the reaction wheels are canted at 45 degree angles. The sensor components are placed at the outer edge of the component plate, such that they have a field of view through holes in the outer bus wall. The sensors face directions in which there are no solar arrays blocking the view. In Figure 17-4, the ADCS components are shown integrated into the structure.

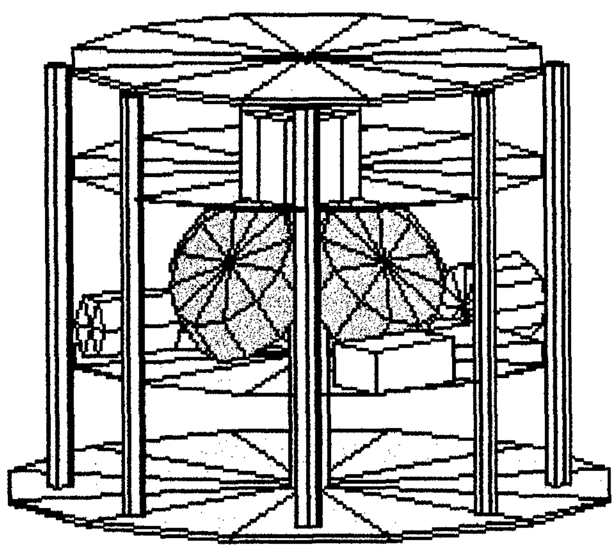
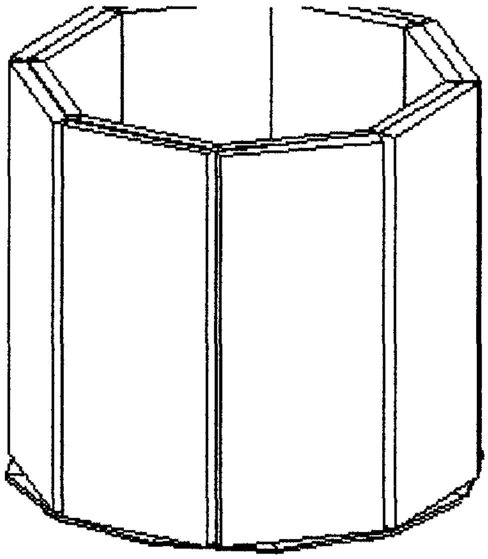


Figure 17-4: MAXTAC ADCS

17.5.4 EPS

Due to the placement of the reaction wheels, the batteries hang from the bottom of the 2nd component plate. The regulator and CPU are placed symmetrically where space allows. The stowed and deployed EPS configurations are shown in Figure 17-5.



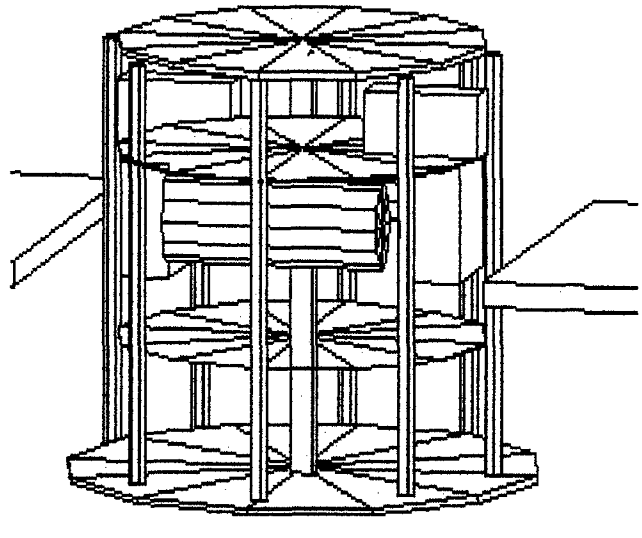


Figure 17-5: MAXTAC EPS

17.5.5 TTC

The SGLS antenna on the third component level is placed close to the outer edge of the bus. This antenna must be deployed on a boom with an appropriate mechanism. TT&C/CPU components are shown in Figure 17-6.

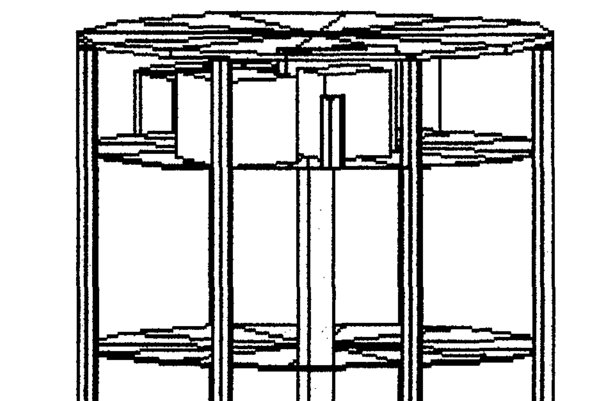


Figure 17-6: MAXTAC TT&C/CPU

17.5.6 MAXTAC SMASH

Figure 17-7 shows the top level of the bus, with the SMASH being utilized by a high data rate communications package.

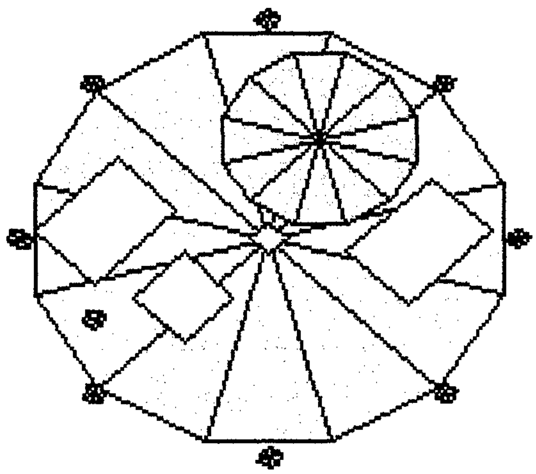


Figure 17-7: MAXTAC SMASH

17.5.7 MAXTAC Composite

The entire MAXTAC design is shown in Figure 17-8. The primary characteristics of MAXTAC are listed in Table 17-42.

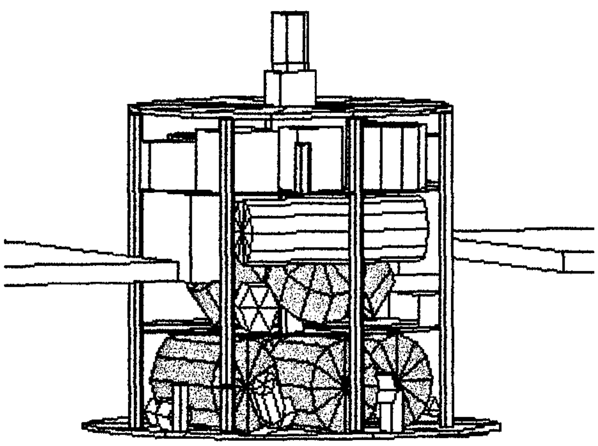


Figure 17-8: MAXTAC (deployed)

Table 17-42: Primary Characteristics of MAXTAC

Mass (kg)	262.5
Height of bus (cm)	71
Average Power (watts)	447
Peak Power (watts)	1000

17.6 Variations on the Baseline

17.6.1 MIDTAC

The MIDTAC alternative is exactly the same as the MAXTAC bus, except that the bottom level is shorter due to the smaller propellant tanks.

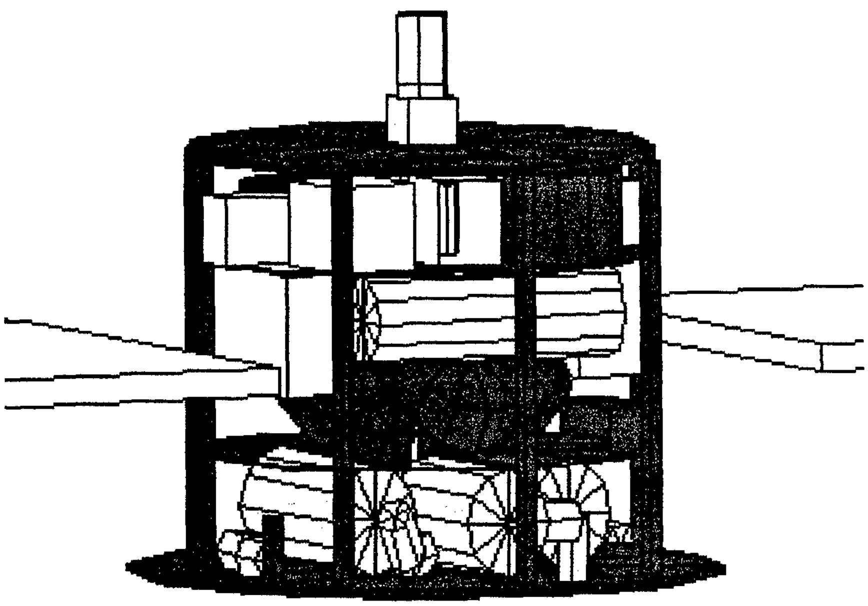


Figure 17-9: MIDTAC (deployed)

Table 17-43: Primary Characteristics of MIDTAC

Mass (kg)	242.9
Height of bus (cm)	68
Average Power (watts)	447
Peak Power (watts)	1000

17.6.2 LOWTAC

The LOWTAC bus is also the same as MAXTAC, but with an even shorter bottom level.

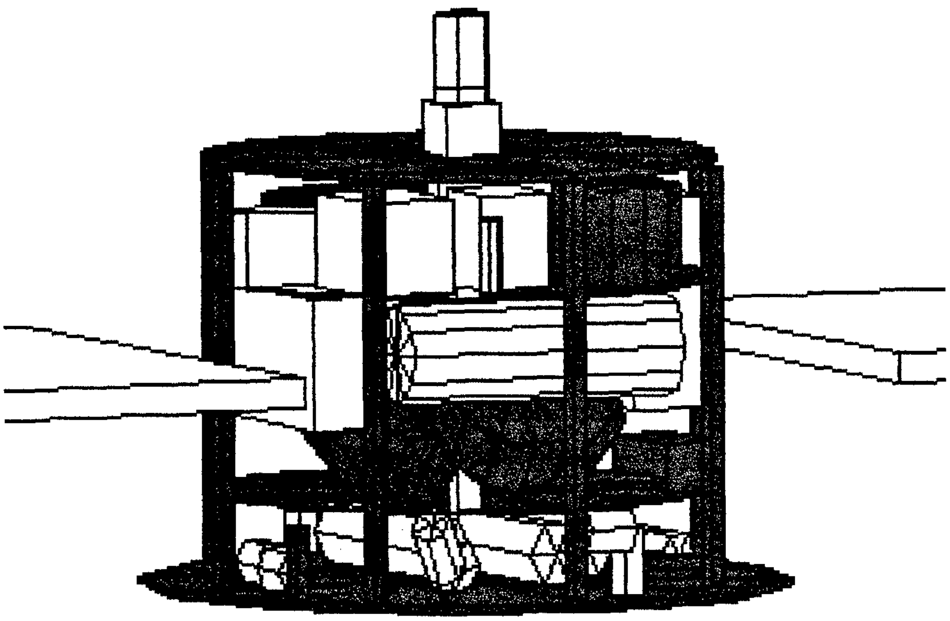


Figure 17-10: LOWTAC (deployed)

Table 17-44: Primary Characteristics of LOWTAC

Mass (kg)	215.7
Height of bus (cm)	63
Average Power (watts)	447
Peak Power (watts)	1000

17.6.3 MACTAC-N

In the next three alternatives, the layout of the bottom level is the same as in the previous three alternatives. However, there is no SMASH. In fact, since the removal of the SMASH space leaves only a few components on the top level, these components were relocated onto the second level to reduce the height of the bus. But the height of the second level was increased in order to fit all of components within.

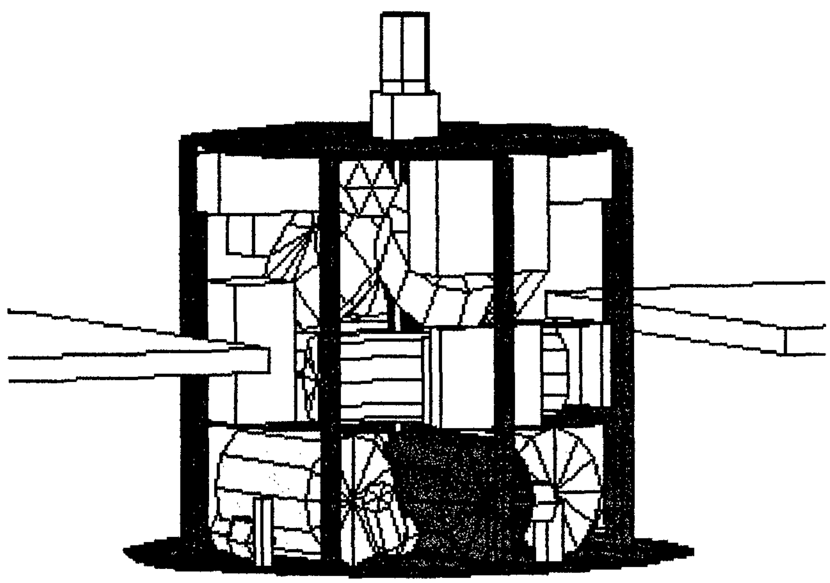


Figure 17-11: MAXTAC-N (deployed)

Table 17-45: Primary Characteristics of MAXTAC-N

Mass (kg)	256.1
Height of bus (cm)	69
Average Power (watts)	447
Peak Power (watts)	1000

17.6.4 MIDTAC-N

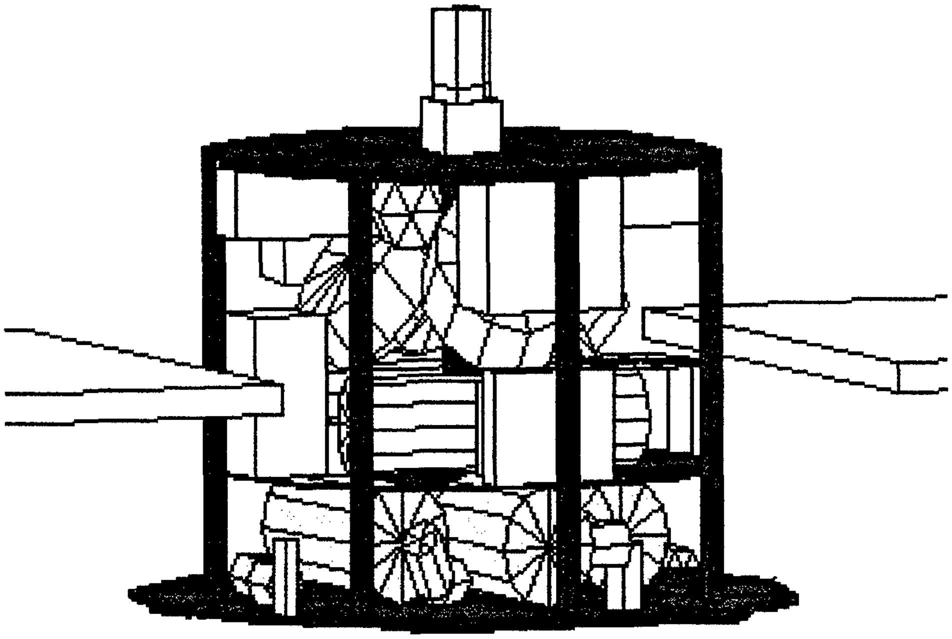


Figure 17-12: MIDTAC-N (deployed)

Table 17-46: Primary Characteristics of MIDTAC-N

Mass (kg)	236.5
Height of bus (cm)	66
Average Power (watts)	447
Peak Power (watts)	1000

17.6.5 LOWTAC-N

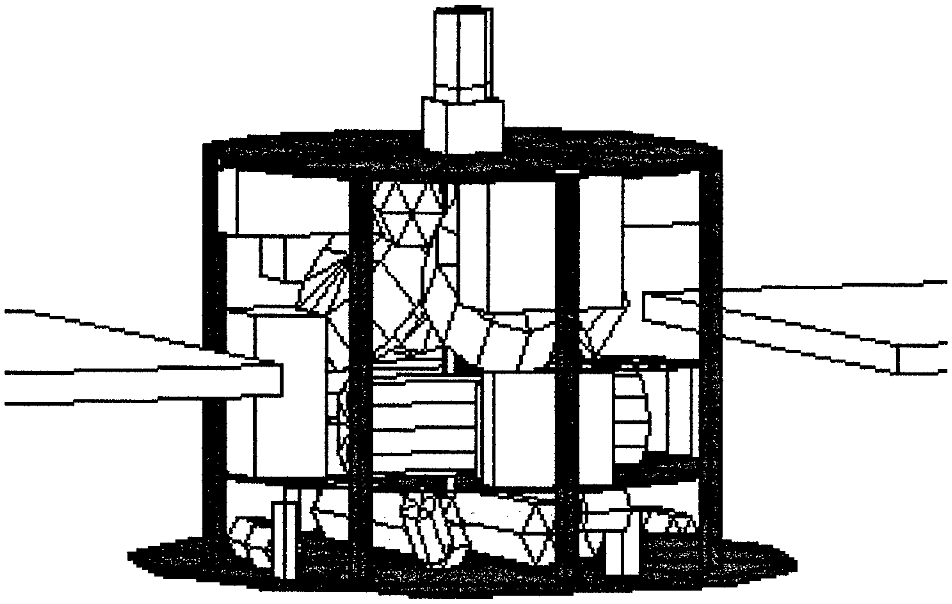


Figure 17-13: LOWTAC-N (deployed)

Table 17-47: Primary Characteristics of LOWTAC-N

Mass (kg)	209.3
Height of bus (cm)	61
Average Power (watts)	447
Peak Power (watts)	1000

17.6.6 MIDTAC-23

Finally, a 23" LV interface version of Modsat was created simply to broaden the solution space somewhat. The MIDTAC platform was chosen, although any of the tactical profiles would have been appropriate. The MIDTAC-23 bus is exactly the same as the MIDTAC bus, except that it is placed higher in the LV due to the interface.

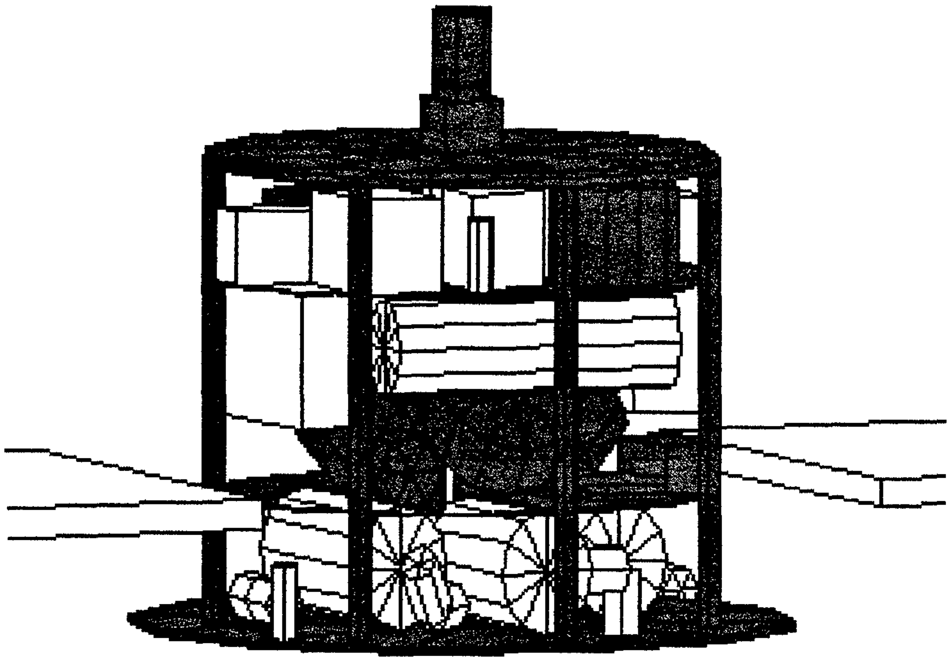


Figure 17-14: MIDTAC-23 (deployed)

Table 17-48: Primary Characteristics of MIDTAC-23

Mass (kg)	292.8
Height of bus (cm)	95.7
Average Power (watts)	390
Peak Power (watts)	950

17.7 Summary

To primary characteristics of each alternative are summarized in , where

1 = MAXTAC
2 = MIDTAC
3 = LOWTAC
4 = MAXTAC-N
5 = MIDTAC-N
6 = LOWTAC-N
7 = MIDTAC-23

Table 17-49: Primary Characteristics of All Designs

Characteristic	1	2	3	4	5	6	7
Mass (kg)	262.5	242.9	215.7	256.1	236.5	209.3	292.8
Height (cm)	71	68	63	69	66	61	95.7
Avg Power (W)	447	47	447	447	447	447	390
Peak Power (W)	1000	1000	1000	1000	1000	1000	950

18. System Analysis

This section explains the method by which the alternative designs were compared and rated. The key factors that the team used to perform the analysis are discussed for each objective. Finally, the results of the analysis are presented.

18.1 Definitions

The alternative designs are classified two ways, as follows.

- Categories: There are two categories, based on the existence of the third level of the bus.
  - ⇒ Category One (Cat 1): Includes all three-level/SMASH designs. These designs are designated as "free space" alternatives, since they have a large, reserved space on the third level. This space is recommended for a dedicated high data rate (HDR) communications package.
  - ⇒ Category Two (Cat 2): Includes all two-level/No SMASH designs. These designs are designated as "no free space" alternatives, since there is no third level with reserved space. Any dedicated HDR communications package must be integrated into the mission module.
- Tactical Profiles: There are three profiles, based on the amount of ΔV provided by the propulsion subsystem.
  - ⇒ Profile One: 100 m/s of ΔV, for little or no plane change capability.
  - ⇒ Profile Two: 300 m/s of ΔV, for 1.5° of plane change at 350 km altitude.
  - ⇒ Profile Three: 450 m/s of ΔV, for 2.5° of plane change at 350 km altitude.

18.2 Method

In the following analysis, the seven design alternatives are analyzed objective by objective. For each objective, the team evaluated the performance of each alternative. For objectives with natural measures of effectiveness, the performance value was obtained from the Modsat model. For objectives with attribute scale MOEs, the team rated the performance of each alternative against its attribute scale. Ratings were given by group consensus, which was reached in each case after thorough discussion. The scores were assigned to each alternative individually; however, the judgments still involved many comparisons between the alternatives. It was discovered that for many objectives, groups of scores could be assigned by category or profile, depending on which classification was relevant to the judgment. Note that the analysis involved only the bottom-level objectives, since only these objectives have MOEs.

For many of the objectives, there was no distinguishable difference between the alternatives. This possibility was discussed in Value System Design. All of the objectives were critical to guiding the design process, and the tradeoffs and decisions were made in accordance with these objectives. As a result of these decisions, the alternative solutions have many similar characteristics, and perform equally well in several of the objectives. But the Modsat evaluation model requires scores for each objective; thus, in such cases, the team often assigned a nominal attribute value of 2.5 to the alternatives.

18.3 Analysis

1. Minimize time to full rate production:

A design with spread-out and accessible components performs well in this attribute. Cat 2 designs are more difficult to manufacture/assemble, since the components are more densely grouped. If a high data rate communications package is required, which is highly likely for all Modsat mission, the Cat 2 configuration will require the mission module package to incorporate additional communications equipment. Thus, Cat 2 designs do not have the flexibility that Cat 1 designs do.

Table 18-1: Min Time to Full Rate Production

	CAT 1	CAT 2	
MAXTAC	3.5	2.5	MAXTAC-N
MIDTAC	3.5	2.5	MIDTAC-N
LOWTAC	3.5	2.5	LOWTAC-N
MIDTAC-23	3.0		

2. Minimize research, development, test and engineering cost (numbers obtained from the Modsat cost model):

Table 18-2: Min RDT&E Cost

Design	Cost ($ million)
MAXTAC	77.50
MIDTAC	72.43
LOWTAC	64.74
MAXTAC-N	75.15
MIDTAC-N	70.10
LOWTAC-N	62.36
MIDTAC-23	70.70

3. Minimize production cost (numbers obtained from the Modsat cost model):

Table 18-3: Min Production Cost

Design	Cost ($ million)
MAXTAC	30.09
MIDTAC	27.00
LOWTAC	21.71
MAXTAC-N	28.93
MIDTAC-N	25.80
LOWTAC-N	20.55
MIDTAC-23	26.71

4. Minimize retirement cost:

The more fuel a satellite has, the more flexible are its options for retirement. Fuel can be used to de-orbit the satellite, or to boost it to a useless orbit. But without this fuel, the satellite can only be retired gradually, through orbital decay. In this case, if the useful life of the satellite has expired, it remains a drain on tracking resources. Thus, the scores for this attribute are roughly proportional to the amount of fuel used in each design. The following scale was developed:

Delta velocity (meters/sec)	Score
450	4.5
300	3.0
100	1.0

Table 18-4: Min Retirement Cost

	CAT 1	CAT 2	
MAXTAC	4.5	4.5	MAXTAC-N
MIDTAC	3.0	3.0	MIDTAC-N
LOWTAC	1.0	1.0	LOWTAC-N
MIDTAC-23	3.0		

5. Minimize cost of telemetry, tracking, and commanding:

Cat 1 alternatives guarantee a space on the bus that can be used for a dedicated HDR package for the user. This package could possibly be used for command and control. This flexibility with regard to the TT&C function gives Cat 1 an advantage over Cat 2, although not much of one. as part of the TT&C function. Thus, Cat 1 designs guarantee more options

Table 18-5: Min Cost of TT&C

	CAT 1	CAT 2	
MAXTAC	3.5	3.0	MAXTAC-N
MIDTAC	3.5	3.0	MIDTAC-N
LOWTAC	3.5	3.0	LOWTAC-N
MIDTAC-23	3.5		

6. Minimize cost of mission module integration and system test:

Cat 1 alternatives perform somewhat better on this attribute than Cat 2 alternatives. Since Cat 2 alternatives do not have the free space, their integration with mission modules is more complicated. This is due to the fact that a HDR package, or any other additional components which could have been placed in the free space, must be integrated within the mission module. One must also consider the additional difficulty of controlling the center of mass when placing additional components in the mission module. Also, the free space in Cat 1 alternatives allows room for additional components. These components can be incorporated into the bus and fully tested prior to mission module integration, thereby reducing the cost of the integration effort. On the other hand, Cat 1 alternatives allow for more variability in the system configuration, which would increase the workload during tests of the integrated satellite. Also, integration and test efforts for

Cat 1 alternatives must take into account the extra equipment and its associated data and power lines.

Table 18-6: Min Cost of Mission Module Integration and System Test

	CAT 1	CAT 2	
MAXTAC	3.75	3.0	MAXTAC-N
MIDTAC	3.75	3.0	MIDTAC-N
LOWTAC	3.75	3.0	LOWTAC-N
MIDTAC-23	3.75		

7. Minimize cost of maintenance:

Alternatives with smaller fuel tanks and better access for removal and replacement of components perform better on this attribute. Thus, volume of fuel and number of component levels are key elements. In Cat 1 designs, the components are less densely packed than in Cat 2 designs. This characteristic, along with the additional plate, allows for easier access to components. It was decided that as the fuel tanks increase in volume and weight, their maintenance becomes more difficult. The tank structures are larger and therefore harder to handle and transport. Moreover, the additional fuel increases the hazards of working with and storing hydrazine.

Table 18-7: Min Cost of Maintenance

	CAT 1	CAT 2	
MAXTAC	4.0	2.5	MAXTAC-N
MIDTAC	4.5	3.0	MIDTAC-N
LOWTAC	5.0	2.5	LOWTAC-N
MIDTAC-23	4.5		

8. Minimize cost of storage, handling, and transportation:

Although Cat 1 and Cat 2 are not vastly different in this objective, the additional complexity of the more compact Cat 2 design may lead to more precautions and procedures in the handling and transporting of Cat 2 spacecraft. The amount of on-board fuel is another consideration. Smaller and lighter propulsion tanks imply less effort in storage, handling, and transportation. Thus, alternatives with smaller tanks and more component plates perform better on this attribute.

Table 18-8: Min Cost of Storage, Handling, and Transportation

	CAT 1	CAT 2	
MAXTAC	3.0	2.5	MAXTAC-N
MIDTAC	3.5	3.0	MIDTAC-N
LOWTAC	4.0	3.5	LOWTAC-N
MIDTAC-23	3.5		

9. Minimize cost of launch integration and test:

Launch integration is not primarily concerned with the internal makeup of the satellite. However, the weight and size of the satellite are key factors. Smaller and lighter satellites perform better on this attribute. Since the Cat 2 design is more compact and shorter than Cat 1. And for a given tactical profile, the Cat 2 alternative is also lighter than the Cat 1 alternative. Thus, it is easier to handle.

Table 18-9: Min Cost of Launch Integration and Test

	CAT 1	CAT 2	
MAXTAC	2.5	3.0	MAXTAC-N
MIDTAC	3.0	3.5	MIDTAC-N
LOWTAC	3.5	4.0	LOWTAC-N
MIDTAC-23	2.5		

10. Minimize preparation time to launch:

Since this objective covers the time it takes a bus to go from storage to launch, the properties of weight, height, fuel tank size, and complexity are the primary considerations. Therefore, the scores from "minimize cost of maintenance" and "minimize cost of launch integration and test" were combined and averaged. Shorter, lighter, and less compact satellites perform better on this attribute.

Table 18-10: Min Prep Time to Launch

	CAT 1	CAT 2	
MAXTAC	3.25	2.75	MAXTAC-N
MIDTAC	3.75	3.25	MIDTAC-N
LOWTAC	4.25	3.75	LOWTAC-N
MIDTAC-23	3.5		

11. Maximize capability for tactical maneuvers:

Alternatives with more fuel perform better on this attribute, since they allow for more ΔV. It should be noted that slew rate is an important factor in tactical capability. However, since all of the alternatives have the same actuator system, slew rate was ignored for this analysis. The following scale was developed:

Delta velocity (meters/sec)	Score
450	4.5
300	3.0
100	1.0

Table 18-11: Max Capability for Tactical Maneuvers

	CAT 1	CAT 2	
MAXTAC	4.5	4.5	MAXTAC-N
MIDTAC	3.0	3.0	MIDTAC-N
LOWTAC	1.0	1.0	LOWTAC-N
MIDTAC-23	3.0		

12. Minimize data latency:

For a given HDR communications package, data latency is not affected by whether this package is placed on the mission module or on the bus. All other factors that could affect data latency are the same for all alternatives. Therefore, a nominal score of 2.5 was assigned to each design.

Table 18-12: Min Data Latency

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

13. Maximize reliability:

All of the designs use the same set of components. Therefore, it was assumed for this preliminary analysis that all designs would have the same reliability. Equal values were entered into the model in the form of a nominal attribute scale number.

Table 18-13: Max Reliability

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

14. Maximize survivability:

The additional size and electrical wiring of the Cat 1 alternatives make them more susceptible to enemy-directed radiation, EMP, etc., than Cat 2 alternatives. However, designs with more fuel are more able to perform evasive maneuvers in order to avoid ASAT attacks. Satellites with more fuel and a more compact frame perform better on this attribute.

Table 18-14: Max Survivability

	CAT 1	CAT 2	
MAXTAC	4.25	4.5	MAXTAC-N
MIDTAC	2.75	3.0	MIDTAC-N
LOWTAC	0.75	1.0	LOWTAC-N
MIDTAC-23	2.75		

15. Minimize cost risk:

The cost estimating relationships used for this study assume the use of standard components and production methods. Cat 2 designs follow the more traditional method of building a satellite. Thus, the team has more confidence in the cost estimates of Cat 2 designs, over the more non-traditional Cat 1 designs.

Table 18-15: Min Cost Risk

	CAT 1	CAT 2	
MAXTAC	2.5	3.5	MAXTAC-N
MIDTAC	2.5	3.5	MIDTAC-N
LOWTAC	2.5	3.5	LOWTAC-N
MIDTAC-23	2.5		

16. Minimize performance risk:

Since all the designs were built with the same components, there is no appreciable difference between them with respect to this attribute. However, more detailed modeling at some point in the future may reveal greater risk in the more non-traditional approach of the Cat 1 design.

Table 18-16: Min Performance Risk

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

17. Minimize schedule risk:

In the Cat 2 configuration, components are placed very close together. This may lead to unexpected problems with construction, integration, and test. Moreover, any unforeseen changes to the design would be extremely difficult to implement, and would require a great deal of re-engineering. Therefore, less confidence should be placed in development/production/test schedules for Cat 2 alternatives, than for Cat 1 alternatives. The roominess of the Cat 1 designs give them an advantage in this area, since changes in the design could be more easily implemented. However, the allowed variation in the configurations of the Cat 1 designs may make them vulnerable to unexpected delays.

Table 18-17: Min Schedule Risk

	CAT 1	CAT 2	
MAXTAC	3.5	2.0	MAXTAC-N
MIDTAC	3.5	2.0	MIDTAC-N
LOWTAC	3.5	2.0	LOWTAC-N
MIDTAC-23	3.5		

18. Maximize pointing accuracy:

Since all of the alternatives have the same suite of sensors an actuators, there is no difference between them with respect to this attribute. Thus, equal values were entered into the model in the form of a nominal attribute scale number.

Table 18-18: Max Pointing Accuracy

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

19. Maximize data storage:

All of the designs allow for two gigabytes of storage. Thus, equal values were entered into the model in the form of a nominal attribute scale number.

Table 18-19: Max Data Storage

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

20. Maximize average mission module power:

For all designs except MIDTAC-23, the same amount of surface area is provided for the solar arrays. MIDTAC-23 has less available height for the stowed arrays; thus, it has less surface area and does not generate as much power. Average available power is shown in Table 18-20 (numbers obtained from Modsat model).

Table 18-20: Max Average Mission Module Power

Design	Power (watts)
MAXTAC	319.2
MIDTAC	319.2
LOWTAC	319.2
MAXTAC-N	319.2
MIDTAC-N	319.2
LOWTAC-N	319.2
MIDTAC-23	261.9

21. Maximize allowable mission module weight numbers obtained from Modsat model):

Table 18-21: Max Allowable Mission Module Weight

Design	Weight (kg)
MAXTAC	40.50
MIDTAC	60.56
LOWTAC	87.75
MAXTAC-N	47.51
MIDTAC-N	67.27
LOWTAC-N	94.76
MIDTAC-23	64.91

22. Maximize adaptability:

With the additional plate and free space in the Cat 1 designs, the satellite integrator has more flexibility in placing additional components on the bus. Alternatives with more fuel are more adaptable to changes in mission profiles.

Table 18-22: Max Adaptability

	CAT 1	CAT 2	
MAXTAC	4.5	3.0	MAXTAC-N
MIDTAC	4.0	2.5	MIDTAC-N
LOWTAC	3.5	1.5	LOWTAC-N
MIDTAC-23	4.0		

23. Maximum orbital accuracy:

All of the alternatives were given enough fuel, in addition to that reserved for tactical maneuvering, to maintain orbit at 350 km for one year. Thus, there is no difference between the alternatives with respect to this attribute. Equal values were entered into the model in the form of a nominal attribute scale number.

Table 18-23: Max Orbital Accuracy

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

24. Maximize data down-link rate:

All of the alternatives could incorporate the same HDR communications package. The location of this package on the satellite should not affect its throughput performance. Thus, there is no difference between the alternatives with respect to this attribute. Equal values were entered into the model in the form of a nominal attribute scale number.

Table 18-24: Max Data Down-link Rate

	CAT 1	CAT 2	
MAXTAC	2.5	2.5	MAXTAC-N
MIDTAC	2.5	2.5	MIDTAC-N
LOWTAC	2.5	2.5	LOWTAC-N
MIDTAC-23	2.5		

25. Maximize peak mission module power:

Refer to the discussion for objective 20, "maximize average mission module power." Numbers in Table 18-25 were obtained from the Modsat model.

Table 18-25: Max Peak Mission Module Power

Design	Power (watts)
MAXTAC	872.2
MIDTAC	872.2
LOWTAC	872.2
MAXTAC-N	872.2
MIDTAC-N	872.2
LOWTAC-N	872.2
MIDTAC-23	814.9

26. Maximize allowable mission module volume (numbers obtained from Modsat model):

Table 18-26 Max Allowable Mission Module Volume

Design	Volume (m^3)
MAXTAC	0.7616
MIDTAC	0.7855
LOWTAC	0.8094
MAXTAC-N	0.7825
MIDTAC-N	0.8005
LOWTAC-N	0.8304
MIDTAC-23	0.7855

27. Minimize thermal transfer:

Since the components in the Cat 1 designs are more spread out, they are less likely to be thermally impacted by neighboring components than are components in the Cat 2 designs. A decrease in the cross-sectional area of the satellite reduces the amount of solar absorption.

Table 18-27: Min Thermal Transfer

	CAT 1	CAT 2	
MAXTAC	3.0	2.0	MAXTAC-N
MIDTAC	3.5	2.5	MIDTAC-N
LOWTAC	4.0	3.0	LOWTAC-N
MIDTAC-23	3.5		

18.4 Results

The main results for each alternative are presented below in Table 18-28, Table 18-29, and Figure 18-1. The scores were obtained from the Modsat model. The scores listed under each main objective in Table 18-28 are weighted scores; that is, they are the result of multiplying the utility scores for a given objective by that objective's weight. For a given alternative, the sum of the weighted scores for each main objective yields the overall utility score in the final column.

Table 18-28 Weighted Scores: Standard Weights

Weight	0.1382	0.3119	0.1527	0.1746	0.2226	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.07329	0.2038	0.1048	0.1131	0.1123	0.6073
MIDTAC	0.09589	0.1835	0.1048	0.1058	0.1309	0.6209
LOWTAC	0.1165	0.1546	0.1048	0.06152	0.151	0.5884
MAXTAC-N	0.07378	0.1962	0.1054	0.1169	0.1134	0.6057
MIDTAC-N	0.09354	0.1763	0.1054	0.0947	0.1296	0.5995
LOWTAC-N	0.1098	0.1469	0.1054	0.06507	0.152	0.5792
MIDTAC-23	0.09937	0.18	0.1048	0.09085	0.09914	0.5742

The final scores for each alternative are presented in again in Table 18-29, where the alternatives have been ranked. The MIDTAC design scores higher than the other alternatives. This comparison is clearer in Figure 18-1.

Many factors combined to give the MIDTAC alternative the highest score. This design fared well in the trade-off between tactical responsiveness and mission utility, the two highest-weighted objectives. In the other three objectives, which are lower-weighted and therefore less influential, MIDTAC scored in the middle of the alternatives. The MAXTAC alternative also fared well, receiving the second highest score. This is primarily due to its tactical responsiveness, by far the highest-weighted objective in this study. The MIDTAC-23 alternative fared the worst of the designs. It did not score near the top for any of the objectives, and received a very low score for mission utility, the second-highest weighted objective of this study. This is because it used a great deal more of the launch vehicle fairing volume, and therefore reduced the area available for the solar arrays.

Table 18-29: Ranking of Alternatives; Standard Weights

1	MIDTAC	0.6209
2	MAXTAC	0.6073
3	MAXTAC-N	0.6057
4	MIDTAC-N	0.5995
5	LOWTAC	0.5884
6	LOWTAC-N	0.5792
7	MIDTAC-23	0.5742

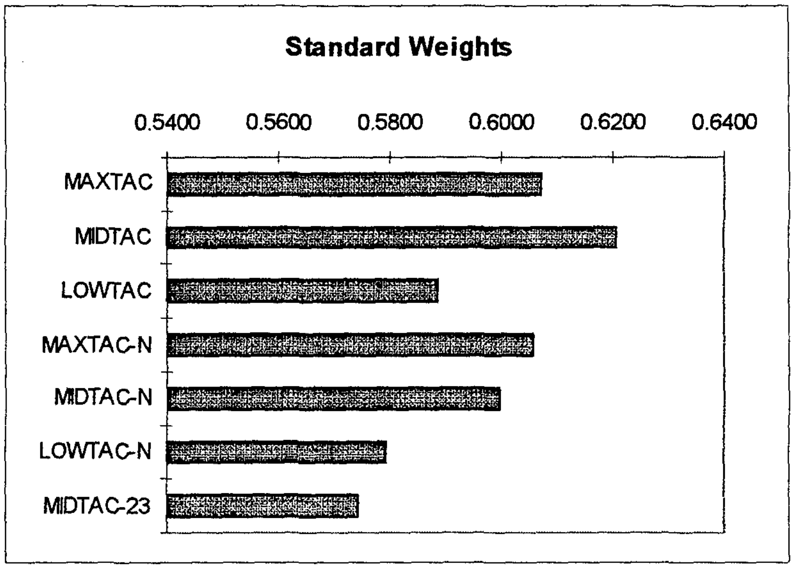


Figure 18-1: Performance at Standard Weights

19. Decision Making

The purpose of this chapter is to provide the chief decision maker with additional information that will aid in the making of decisions regarding the implementation of this program. The results of the previous chapter revealed that the MIDTAC alternative appears to be the best solution. However, these results are dependent on several factors, the chief factor being the values of the priority weights for the objectives. It should be noted that these priority weights were determined through subjective judgments. As mentioned earlier in the study, these judgments were submitted at a given time, under a given set of technological, political, and other environmental factors. Thus, the analysis discussed in this chapter was performed in order to demonstrate the sensitivity of the results to changes in the objective weights, due to shifting environmental factors.

19.1 Sensitivity Analysis

In this analysis, environmental scenarios were created in order to examine the effect of changes in the top-level objective weights. In each scenario, the weight of one of the objectives was increased to correspond with a certain environmental situation. The weights of the four remaining objectives were scaled down proportionally, so that all of the weights would still sum to one. With the new weights, the overall utility function was performed on each of the alternatives, and the results were recorded. In some cases, an alternative other than MIDTAC received the highest score. The first five scenarios were created by multiplying the weights of one of the five top-level objectives by two, and rescaling the other four objective weights accordingly. In the final "extreme cost" scenario, the cost objective was given a weight of 0.5.

19.1.1 Cost Scenario

In the original set of weights, the minimization of cost was given relatively low priority. It is conceivable that this objective could become much more important due to cuts in the budget and other funding problems. Therefore, the cost weight was doubled to 0.2764. Table 19-30 and Figure 19-1 below demonstrate the performance of the alternatives under this scenario. MIDTAC still received the highest score. In this scenario, cost and responsiveness are the two highest-weighted objectives, and the MIDTAC alternative optimizes the trade-off between these two factors. The LOWTAC alternative replaced MAXTAC with the second highest score. This is not surprising, since MAXTAC receives a low utility score for cost, which is more heavily weighted in this scenario.

Table 19-30: Weighted Scores; Cost x 2

Weight	0.2764	0.2688	0.1316	0.1505	0.1918	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.1466	0.1711	0.08802	0.09494	0.09432	0.5950
MIDTAC	0.1918	0.1541	0.08802	0.08885	0.1099	0.6327
LOWTAC	0.2331	0.1298	0.08802	0.05165	0.1268	0.6294
MAXTAC-N	0.1475	0.1647	0.08848	0.09817	0.09519	0.5940
MIDTAC-N	0.1871	0.148	0.08848	0.07952	0.1088	0.6119
LOWTAC-N	0.2195	0.1233	0.08848	0.05464	0.1277	0.6136
MIDTAC-23	0.1987	0.1511	0.08802	0.07628	0.08324	0.5973

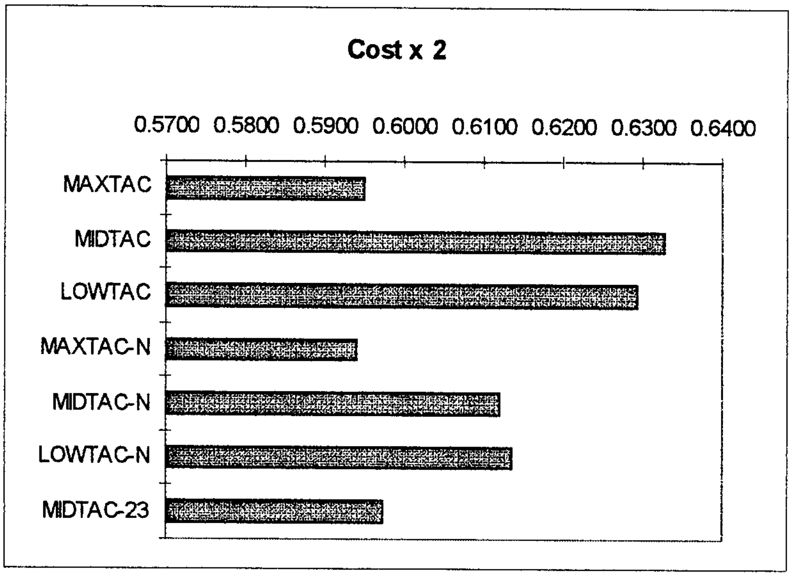


Figure 19-1: Performance at Cost x 2

19.1.2 Responsiveness Scenario

Tactical responsiveness may become more important due to a proliferation of situations and conflicts around the world where US interests are threatened. In this scenario, the weight of the responsiveness objective was doubled to 0.6238. Table 19-31 and Figure 19-2 below show how the alternatives performed. It is not surprising that the two MAXTAC alternatives received the highest scores, since they were specifically designed for responsiveness. Within each category (see the previous chapter for a definition of categories), the scores decrease with less propulsion capability.

Table 19-31: Weighted Scores; Responsiveness x 2

Weight	0.0951	0.6238	0.1051	0.1201	0.1532	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.04007	0.4075	0.05732	0.06182	0.06142	0.6281
MIDTAC	0.05243	0.367	0.05732	0.05785	0.07154	0.6061
LOWTAC	0.06372	0.3092	0.05732	0.03363	0.08257	0.5464
MAXTAC-N	0.04034	0.3923	0.05762	0.06393	0.06199	0.6162
MIDTAC-N	0.05114	0.3526	0.05762	0.05178	0.07087	0.5840
LOWTAC-N	0.06002	0.2937	0.05762	0.03558	0.08312	0.5300
MIDTAC-23	0.05433	0.3599	0.05732	0.04967	0.05421	0.5754

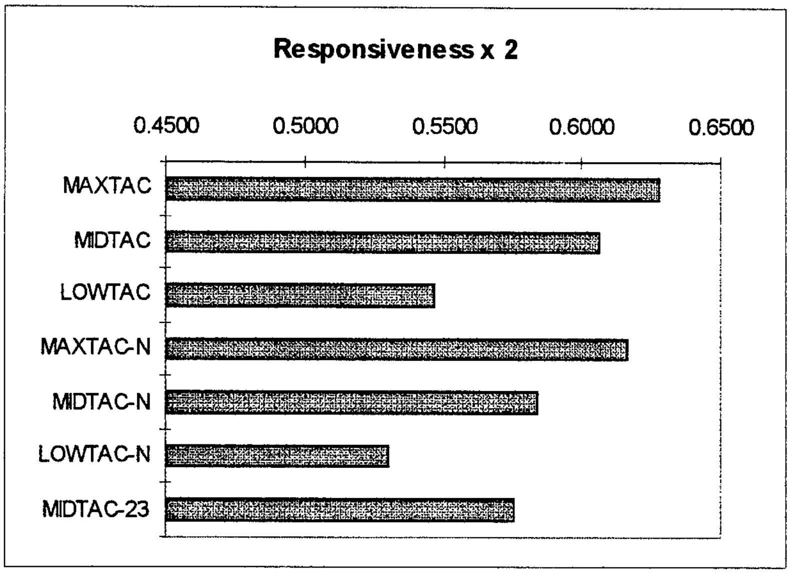


Figure 19-2: Performance at Responsiveness x 2

19.1.3 Risk Scenario

Any number of budgetary, schedule, and performance-related constraints and limitations may combine to give the minimization of cost, schedule, and performance risk a higher priority. For instance, the space program could experience a series of expensive and spectacular failures, which would tend to make program managers much more averse to risk. In this scenario, the weight of the risk objective was doubled to 0.3054. Table 19-32 and Figure 19-3 below show how the alternatives performed. Again, the MIDTAC and MAXTAC alternatives received the highest score. Since the weighted risk scores were and still are equal for the designs within a category, and since there is little difference between these scores across categories, it is not surprising that this scenario does not change the ranking of the alternatives. However, note that the Cat 2 designs score somewhat higher in this scenario, since they perform better in the risk objective than the Cat 1 designs.

Table 19-32: Weighted Scores; Risk x 2

Weight	0.1171	0.26427	0.3054	0.14794	0.18861	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.0593	0.1649	0.2158	0.09149	0.09089	0.6224
MIDTAC	0.07759	0.1485	0.2158	0.08562	0.1059	0.6334
LOWTAC	0.09431	0.1251	0.2158	0.04978	0.1222	0.6072
MAXTAC-N	0.0597	0.1587	0.217	0.09461	0.09174	0.6218
MIDTAC-N	0.07569	0.1427	0.217	0.07663	0.1049	0.6169
LOWTAC-N	0.08882	0.1189	0.217	0.05265	0.123	0.6004
MIDTAC-23	0.0804	0.1456	0.2158	0.07351	0.08022	0.5955

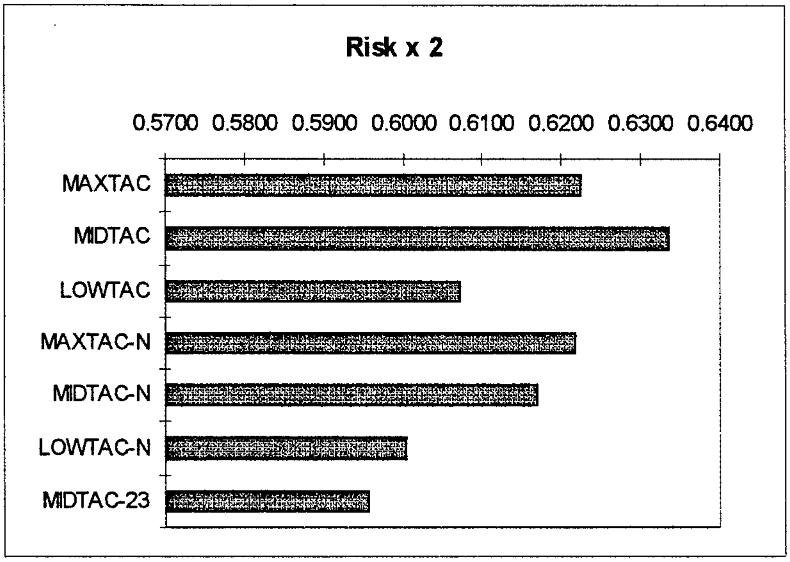


Figure 19-3: Performance at Risk x 2

19.1.4 Availability Scenario

Availability could become more important for several reasons. Air Force doctrine could call for more space support of long-term regional conflicts and peacekeeping missions (such as the Middle East, Bosnia, etc.), in which case high system lifetimes, reliability, and survivability would be an asset. Or a leader in the chain of command over the Modsat program could adopt a conservative stance and request all programs to focus more on reliability and survivability. In this scenario, the weight of the availability objective was doubled to 0.3492. Performance of the alternatives is shown in Table 19-33 and Figure 19-4. MAXTAC-N, MIDTAC, and MAXTAC received the top scores, in that order. Note that MAXTAC-N has the highest score under the availability objective.

Table 19-33: Weighted Scores; Availability x 2

Weight	0.11407	0.25744	0.12604	0.3492	0.18373	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.05778	0.1607	0.08265	0.2262	0.08857	0.6159
MIDTAC	0.0756	0.1447	0.08265	0.2116	0.1032	0.6178
LOWTAC	0.09189	0.1219	0.08265	0.123	0.1191	0.5385
MAXTAC-N	0.05817	0.1547	0.08309	0.2339	0.08939	0.6193
MIDTAC-N	0.07375	0.139	0.08309	0.1894	0.1022	0.5874
LOWTAC-N	0.08655	0.1158	0.08309	0.1302	0.1199	0.5355
MIDTAC-23	0.07835	0.1419	0.08265	0.1817	0.07817	0.5628

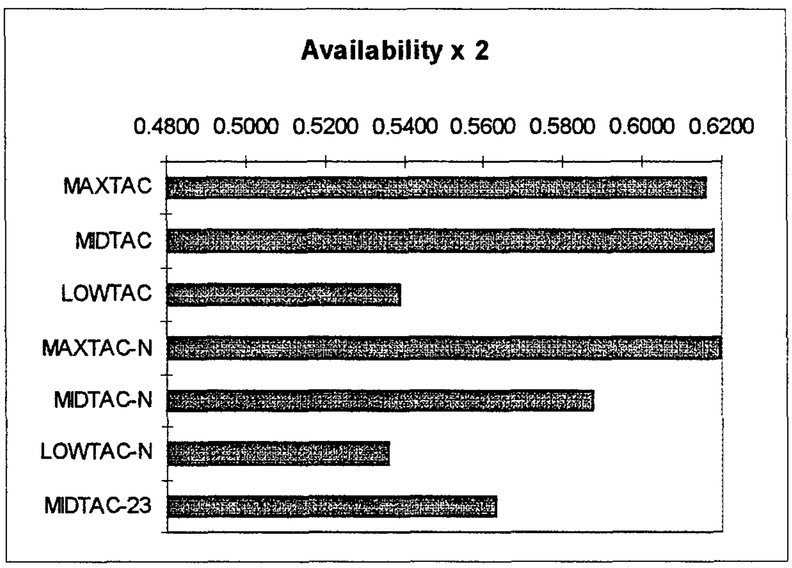


Figure 19-4: Performance at Availability x 2

19.1.5 Utility Scenario

Many factors could combine to give mission utility a higher priority. The standardized approach could become quite appealing to mission program managers, making Modsat a sought-after platform. In this case, maximum mission functionality would become even more important. In this scenario, the weight of the mission utility objective was doubled to 0.4452. The alternatives performed as shown in Table 19-34 and Figure 19-5. LOWTAC received the highest score, while MIDTAC placed second. In general, the smaller satellites did well in this scenario. This is primarily due to their allowance for more weight and volume in the mission module, these attributes being key sub-objectives under mission utility.

Table 19-34: Weighted Scores; Utility x 2

Weight	0.10744	0.24247	0.11871	0.13573	0.4452	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.0523	0.1454	0.07481	0.08069	0.2247	0.5779
MIDTAC	0.06843	0.131	0.07481	0.07551	0.2617	0.6115
LOWTAC	0.08317	0.1103	0.07481	0.0439	0.3021	0.6143
MAXTAC-N	0.05265	0.14	0.0752	0.08344	0.2268	0.5781
MIDTAC-N	0.06675	0.1258	0.0752	0.06758	0.2593	0.5946
LOWTAC-N	0.07834	0.1048	0.0752	0.04644	0.3041	0.6089
MIDTAC-23	0.07091	0.1284	0.07481	0.06483	0.1983	0.5373

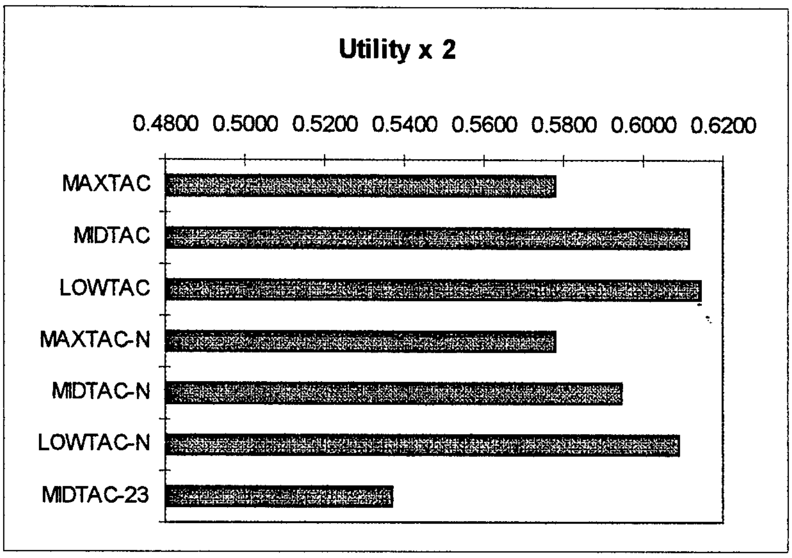


Figure 19-5: Performance at Utility x 2

19.1.6 Extreme Cost Scenario

This scenario was created to examine the effect of placing extreme importance on minimizing cost. This environment could exist in the case of a long term budget crisis, or a severe down-sizing of the military. In general, the priority placed on the cost objective is the most susceptible to change among the top-level objectives. Therefore, it was decided that this investigation would add value to the sensitivity analysis. In this scenario, the cost objective was given a weight of 0.5. The performance of the alternatives is shown in Table 19-35 and Figure 19-6. The LOWTAC alternative received the highest score. This design dominated in performance of the cost objective, due to its small size and low cost of pre-launch operations.

Table 19-35: Weighted Scores; 50 % Cost

Weight	0.5	0.19905	0.09745	0.11143	0.14206	
	Cost	Responsiveness	Risk	Availability	Utility	Total
MAXTAC	0.2651	0.1182	0.06082	0.0656	0.06517	0.5750
MIDTAC	0.3469	0.1065	0.06082	0.06139	0.07592	0.6515
LOWTAC	0.4216	0.08971	0.06082	0.03569	0.08762	0.6954
MAXTAC-N	0.2669	0.1138	0.06114	0.06784	0.06578	0.5755
MIDTAC-N	0.3384	0.1023	0.06114	0.05494	0.0752	0.6320
LOWTAC-N	0.3971	0.08522	0.06114	0.03775	0.08821	0.6694
MIDTAC-23	0.3595	0.1044	0.06082	0.05271	0.05752	0.6350

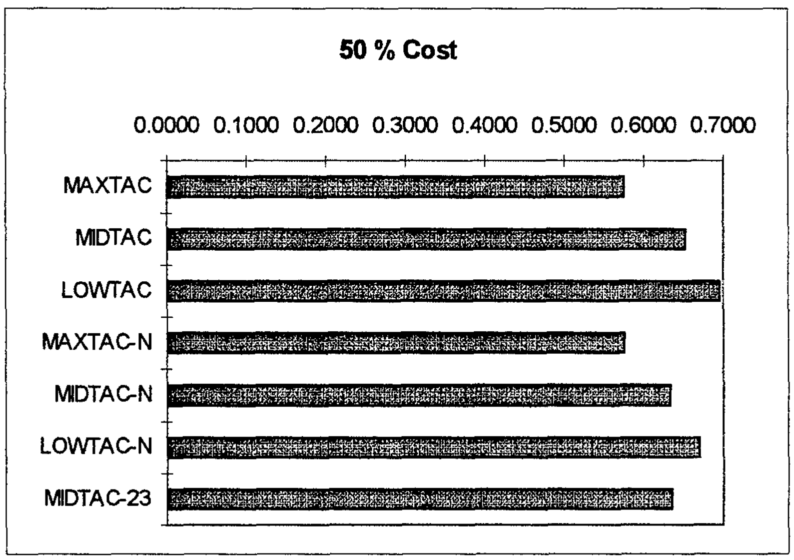


Figure 19-6: Performance at 50 % Cost

19.2 Conclusion

The rankings of the alternatives, for all scenarios, are shown below in Table 19-36. The MIDTAC alternative is the best solution. It scored in the top three in every scenario, with three first place finishes, two second place finishes, and two third place finishes. The ranks for each alternative, summed over all the scenarios, are calculated in Table 19-37. These results further portray MIDTAC as the superior alternative. It is a well-rounded design that optimizes the tradeoffs between the objectives. Its performance in both the responsiveness and utility objectives, the two most critical attributes of the study, contributes to its high score. Note that MIDTAC scores well in both the responsiveness and utility scenarios, while the other alternatives fail to do so.

Table 19-36: Sensitivity Analysis; All Environments

Rank	Standard	Cost x 2	Responsiveness x 2	Risk x 2	Availability x 2	Utility x 2	50% Cost
1	MIDTAC	MIDTAC	MAXTAC	MIDTAC	MAXTAC-N	LOWTAC	LOWTAC
2	MAXTAC	LOWTAC	MAXTAC-N	MAXTAC	MIDTAC	MIDTAC	LOWTAC-N
3	MAXTAC-N	LOWTAC-N	MIDTAC	MAXTAC-N	MAXTAC	LOWTAC-N	MIDTAC
4	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-N	MIDTAC-23
5	LOWTAC	MIDTAC-23	MIDTAC-23	LOWTAC	MIDTAC-23	MAXTAC-N	MIDTAC-N
6	LOWTAC-N	MAXTAC	LOWTAC	LOWTAC-N	LOWTAC	MAXTAC	MAXTAC-N
7	MIDTAC-23	MAXTAC-N	LOWTAC-N	MIDTAC-23	LOWTAC-N	MIDTAC-23	MAXTAC

Table 19-37: Sum of Rankings for the Alternatives

Alternative	Calculation	Sum
MAXTAC	2+6+1+2+3+6+7	27
MIDTAC	**1+1+3+1+2+2+3**	**13**
LOWTAC	5+2+6+5+6+1+1	26
MAXTAC-N	3+7+2+3+1+5+6	27
MIDTAC-N	4+4+4+4+4+4+5	29
LOWTAC-N	6+3+7+6+7+3+2	34
MIDTAC-23	7+5+5+7+5+7+4	40

20. Implementation

This section is intended to aid the CDM in the implementation of the results of the Modsat study. As the purpose of the study was to perform high-level systems engineering, the continuation of the Modsat program will require much more detailed design effort. Moreover, the scope of this study was limited to the design of a spacecraft bus. Many factors and functions must eventually be considered and designed in support of Modsat operations. Recommendations from the team are included summarized as an overall "concept of operations," and are discussed below.

20.1 Continued Design Effort

The systems engineering process is iterative and converging in nature. The scope of each iteration of the design process depends on the current stage within the life-cycle of the program. As the life-cycle progresses, the design process become more detailed. Eventually, the effort converges on an accepted detailed design.

The design information included in this study is relevant for the first stage of the potential life-cycle of Modsat. In this stage, sometimes called "concept exploration," the systems engineer "identifies all reasonable system alternatives that may satisfy the mission need and makes recommendations...; the [CDM] then selects those alternatives or concepts which meet [the] objectives" (Systems Engineering Management Guide, 1989:2-4). If the Modsat program is to progress further, the CDM must build on the concepts and recommendations included in this study.

In the next iteration of the design process, engineers must revisit the selection of components for Modsat, with a view toward optimizing the MIDTAC design. Interfaces must be designed, at the subsystem and component level. In particular, command, telemetry, power, and other signal flows must be examined. The design of software must begin, within the context of the modular satellite operating system recommended by this study. Prototype hardware should be developed to demonstrate the functionality of some of the unique aspects of Modsat, such as its cage structure, or its wrap-around modular solar array assemblies.

At the system level, the mass properties (center of mass, inertia matrix, etc.) must be carefully examined for their effect on stability and control. Control logic should be developed to model the attitude control function of Modsat. Thermal characteristics should be modeled in more detail, and a thermal control system should be designed.

The continued systems engineering effort on Modsat must incorporate concurrent engineering, wherein current engineering efforts reflect consideration of manufacturing, testing, logistics, operational support, etc.

The items mentioned above are just a few of the many challenges awaiting the further design of Modsat.

20.2 Concept of Operations (CONOPS)

The design of a satellite comprises one of the many engineering efforts necessary for the operation of a complete space system architecture. The full architecture encompasses not only the design of the spacecraft and its mission-specific equipment, but also the ground segment (equipment and personnel), the launch segment (equipment and personnel), and the information/ communications architecture (user interface with the system).

20.2.1 Spacecraft Architecture

Implementation of the small tactical satellite design (Tacsat) should take into account the fact that the "baseline" design is generally "over powered" (i.e., the baseline design provides too much average and peak power levels required for operation of most payloads). With this consideration in mind, application of an alternative design architecture (other than the "one size fits all" or "baseline" architecture) becomes desirable for optimization of the bus to the wide variety of mission modules, the majority of which do not require an average power of more than 400 watts or a peak power of over 800 watts. The payloads which may require these high average and peak power loads are those of the active type (LASERs and SARs), whereas passive sensors rarely require more than 100 watts of power (peak -- during a sensing pass). The optimization payoff can be measured in these cases with more available volume on the bus (for other equipment) and less spacecraft bus mass (increasing available mass for the mission module, or allowing a different orbit configuration -- including a higher initial altitude).

Options to the spacecraft designer include either staying with the baseline design for all applications, standardizing the structure and modularizing the power systems, or designing slightly different buses for either active or passive sensor mission modules (a "family" of tactical buses). Given the non-generic nature of its design, a "specialist" or "microsat " architecture, in which the bus is fully optimized to a particular mission module type, falls outside the solution space, which has evolved since Phase I of this study. A similar non-player for the implementation of the MIDTAC, but on the opposite of the specialist architecture, is the original "amoeba" architecture, which would be too generic a concept. It would be unnecessary to have a fully modular bus design in the case of a tactical satellite -- the modularity would evolve into a standard type, defeating the original purpose of adaptability. Adaptability within the scope of tactical space applications is desirable, however. All of these options will meet the needs of the tactical space user and planner.

Tradeoffs involved with the decision to use one particular architecture over the others include the cost involved with the development of "modular" battery assemblies and solar arrays; development and construction cost savings realized by having no modularity (i.e., sticking with the baseline design), the further development costs of additional bus designs (albeit very similar buses with the exact same components in most cases); and operational effectiveness considerations (e.g., time required for integration of the mission module, reliability issues with modular components, changing needs/adaptability).

Another consideration when deciding upon the particular satellite architecture will be the benefit of having small, capable platforms ready to support specific experimental payloads, technology demonstrations, or other (presently undetermined) mission modules (i.e., which approach will be more adaptable for future missions beyond the currently modeled types?). This consideration becomes significant if this capability for adaptation is desirable to the user and planner, as well as other potential DOD users outside the tactical space community who may have a need for a "ride" into space for one or more of their payloads.

Roughly estimating the effectiveness of the architectures, previous data (from the first evaluation of the bus architectures -- see Volume II, Phase 1, System Evaluation) indicate that either the multiple bus architecture ("family") or the modular bus architecture (an infusion of aspects of the original "amoeba" satellite design to the baseline design) would be the optimal choice. The "family" architecture would include two standard buses. The first bus, designed with a lower-power EPS, would be designated to carry the passive sensor mission modules. The second, with the standard MIDTAC power capability, would carry the active sensor mission modules. The MIDTAC bus, evaluated as the best bus design among the standard bus alternatives, includes battery assemblies and solar arrays located in easy-access positions, so a modular design for the battery and solar array interfaces (for varying sizes of either) is also a feasible option. This single subsystem (EPS) and these two primary components (solar arrays and batteries) constitute the modular design points required to (as a first-order estimate) optimize the MIDTAC bus for multiple space roles while using the same exact design; therefore, the tradeoff for these (family or modular) architectures becomes the cost of making the components and their interfaces modular or "hardwiring" the two family buses.

Given the fact that the solar arrays for MIDTAC are already a modular design tailorable to specific needs, and the fact that batteries already come in varying sizes for differing capacity requirements, the modular option should be the architecture chosen for initial implementation of the Modsat. Although actual operational configurations (stored and ready for use) for the Modsat will probably mimic the "family" architecture by including "pre-integrated" buses already tailored for either the low-power or high-power mission modules (possibly already integrated with the some mission modules, as well), the modular design of the Modsat provides ultimate reconfigurability to suit changing needs.

20.2.2 Launch Segment

A small air-launched system provides maximum flexibility for the choice of initial orbit for small tactical spacecraft, because of the fact that any launch azimuth may be chosen. This capability also minimizes the time to reach a chosen target (i.e., the direct approach will be the fastest). The *Pegasus* air launched booster currently represents the only launcher in this category. The *Taurus* booster, while not air launched, is more powerful and was developed specifically for rapid deployment, integration, and launch from unimproved areas, making it also a good choice for the launch segment.

The force structure recommended for employment of small tactical satellite designs consists of elements from the 30th, 45th, and 50th Space Wings (30th, 45th, and 50th SW) working in concert. The 1st Tactical Space Launch Squadron (1st TSLS), based at Vandenberg AFB, CA (30th SW) would be responsible for westward (retrograde and Sun-synchronous) launches over the Pacific Ocean . Similarly the 2nd TSLS (45th SW, Patrick AFB, FL)would be responsible for flights east, over the Atlantic. The 19th Space Operations Squadron (19th SOPS), based at Falcon AFB, CO (50th SW) would have responsibility for Modsat launch and early orbit support, Modsat command and control, Tactical Network (TacNet) maintenance and support, and manning and operation of the Consolidated Tactical Space Control Center (CTSSC -- see below) for in-theatre operations support. Under the direction of a Launch Director, each of the Tactical Space Launch Squadrons would operate B-52 or other *(Pegasus)* carrier aircraft. In addition, a spacecraft-to-mission module integration (SMI) crew, a spacecraft-to-launcher integration (SLI) crew, a loading crew, a flight and launch crew, an analysis crew and a launch (command and control) crew (under the direction of an Operations Director at the 19th SOPS and possibly deployed to a mobile or in-theatre site), would accomplish the integration, loading, flight (to launch location over either ocean), launch, and early orbit support for the Modsat mission.

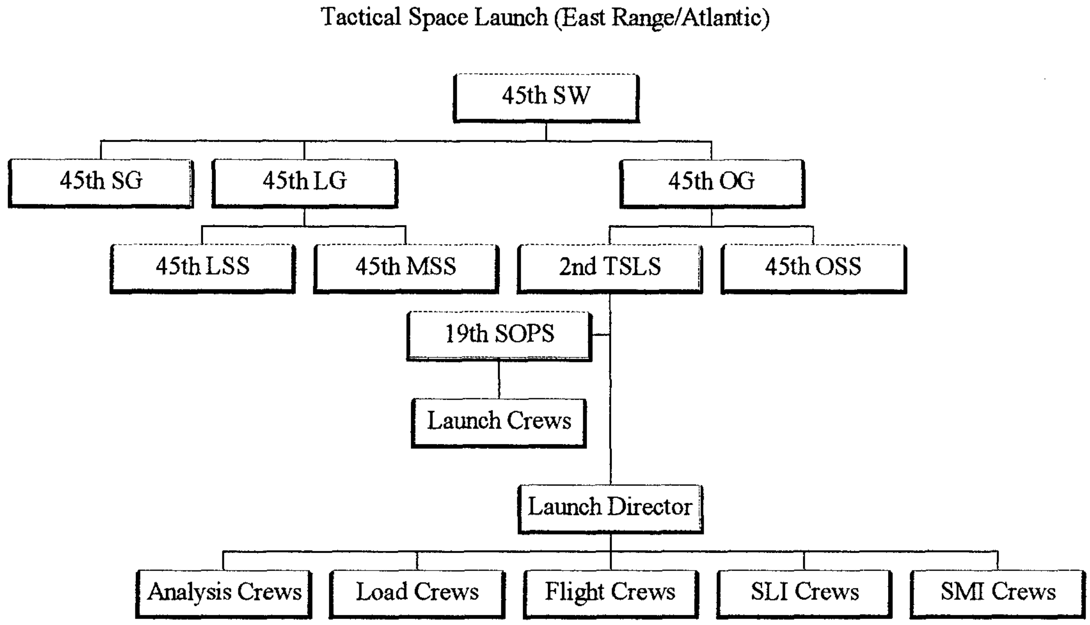


Figure 20-1: Proposed Organizational Structure for Tactical Spacelift (Eastern Range/Atlantic)

The environment for the design study also assumes that a ready supply of launchers, satellite buses, and mission modules is on hand for both rapid response situations and space asset replenishment. Along with the supplies of hardware, other assumptions include ample logistics equipment (transports, "clean" environments, maintenance equipment, special tools, etc.), maintenance personnel, and storage facilities. Specific personnel for each crew include enlisted-grade crew members and technicians/specialists led by officer-grade flight commanders and deputies (doubling as bus, mission module, and launch vehicle experts for their respective crews), and civilian engineering and analysis personnel (in-depth engineering-intensive positions should be made civilian contractor or GS billets to retain "corporate knowledge" on the systems).

Tactical Space Launch (West Range/Pacific)

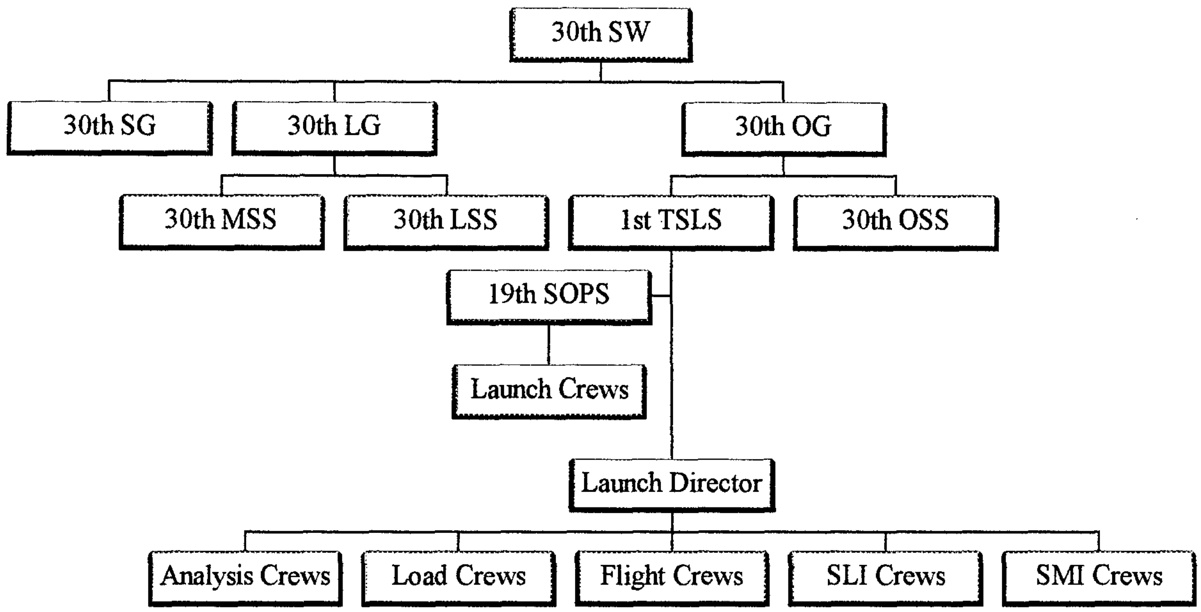


Figure 20-2: Proposed Organizational Structure for Tactical Spacelift (Western Range/Pacific)

20.2.3 Ground Segment and Information/Communications Architecture

The main ground segment for the small tactical satellite should be an X, Ka, or Ku band receiving station, perhaps similar to the Air Force's "Eagle Vision" mobile, in-theatre ground station, which receives mission data directly from both LANDSAT and SPOT satellites, processes the imagery, and overlays the high-resolution visible-region imagery of SPOT with the multispectral imagery of LANDSAT. Multispectral imagery products from Eagle Vision have received rave reviews from operational and theatre commanders (Veseley, 1996). Initial operations testing and/or proof of concept testing (for the new tactical satellites) could be set up to utilize current Eagle Vision equipment.

This central receiving, processing, and distribution station, known as the Consolidated Tactical Space Control Center (CTSCC) would be manned and operated by crews from the 19th SOPS (50th SW) and would also incorporate an S band (SGLS/AFSCN compatible) commanding capability for spacecraft specific commanding. CTSSC terminal stations (including a permanent CTSSC at Falcon AFB, CO) would be deployed at several positions on the Earth to ensure full support for any and every theatre of operations. The CTSSC would be the centerpiece for a tactical space information network into which tactical users would input requests and receive information products. User requests/data updates would be transmitted via wireless ethernet protocols (adapted from currently existing internet protocols) carried through either MILSTAR MDR, Teledesic, Iridium, or a similar high data rate, global communications system.

The process would be as follows:

1) User signs on, and requests are encrypted and transmitted from a laptop or similar small computer in the field or from the cockpit.

2) Requests are received, authenticated, and prioritized (set by the Theatre or Operational Commander) by CTSCC server.

3) CTSCC server queries (via intelligent agents) the types and availabilities of appropriate TACSATs (based on the type(s) of information requested by the user) as well as orbit predictions for those required assets which are available.

4) The server calculates time over target, time to receive mission data, time to process, filter, and package, and time to transmit requested information to the user.

5) If the requested information is available, the CTSSC server replies with the product; if not available, the CTSSC server replies with an estimated time of arrival (ETA) for the product, based on its calculations in 4).

These tasks and the artificially intelligent systems and software required to accomplish them are possible with current computer and software technology and may be made functionally "modular" (in both hardware and software design) to ensure future upgrade capability. To make this system work well for an easily estimable "many" users, the communications systems employed and/or utilized will have to support the necessary bandwidth to successfully support the tactical user in a timely fashion; otherwise, the CTSCC's tactical support capability will be degraded. Centralizing the processing power of this tactical space information system in the CTSSC (as opposed to performing the data-to-information processing on the satellites) allows more processing power to be utilized, allows easy upgrades to software, allows simpler troubleshooting of software, retains the satellites' simplicity -- both from the hardware and the software standpoints, retains central control authority for information dissemination, promotes optimal tasking of resources (again, due to centralized control and prioritization), simplifies communications to and from the satellites (multiple channels are unnecessary), and allows future upgrades to the overall architecture to perhaps "evolve" into a system which incorporates more "on board" processing and direct downlinking to users. Having the information processed by the ground segment is the simpler, more powerful choice for current technology.

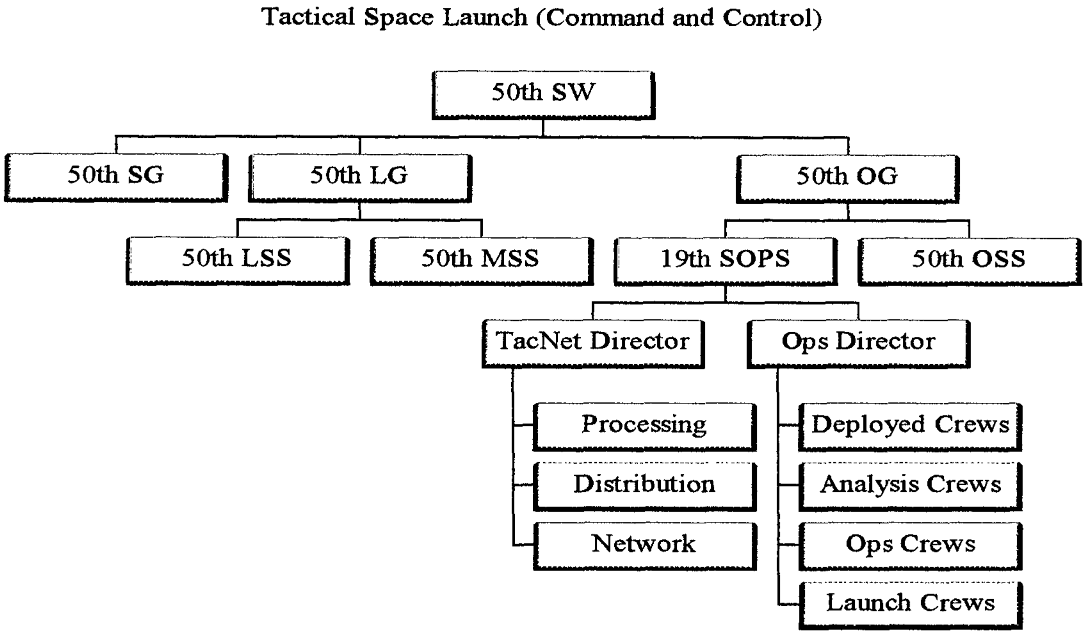


Figure 20-3: Proposed Organizational Structure for Tactical Spacelift (Command and Control)

An alternative to the (assumed baseline) ground-based CTSSC would be a dedicated command and control aircraft, or the CTSCC could be palletized and flown aboard a C-17 or C-5, further enhancing its mobility.

21. Future Technologies and Continuing Investigations

The preliminary design for a small tactical spacecraft bus provides a solid foundation for further investigations as well as continued expansion of the Modsat computer design model. Design efforts generated a tremendous amount of information in addition to producing the Modsat computer-based design tool and a design for a standard, tactically applicable satellite bus. Much of this information was not included within the formal system process either due to the planning horizon for the proposed design (five years) or because of the "nonessential" nature of much of the information (to the design). These topics nonetheless sparked many discussions during meetings and comprise an interesting set of ideas and technologies on which to base a possible future design study (or design studies). The design process also generated several concepts for future scientific research, operational analysis, and system design.

21.1 Modsat Computer Model Enhancements

The Modsat computer model, which was developed to aid in the design and evaluation of small tactical satellite designs, provides a foundation for further enhancements, additions, refinements, and detail. Modsat's coding, though large in scope, is rather simple in style, and it is thoroughly documented internally. Future additions or modifications to the code may include:

- more integration of orbit analysis features
- additional mission module configurations and/or types
- additional cost models

- more detailed component or subsystem modeling
- larger libraries of selectable components
- future technology modeling
- more in-depth launch vehicle modeling and/or design
- more detailed interface modeling

Detailed discussion of the individual sections and functions (background, scope, functionality, limitation(s), and future feature suggestions) of the Modsat computer model can be found in Volume III in the Modeling section.

21.2 Constellation Design

Multiple satellites, carrying active sensors, may in the future be employed in concert as an "array" of sensors. The operational challenge of such a constellation can be solved either by maintaining relative satellite positions and attitudes to within close tolerances (within a few millimeters), or by maintaining extremely accurate position and orientation knowledge (to within a few millimeters). The second solution is the more feasible of the two, because satellites equipped with multiple GPS receivers can produce these knowledge products to within the accuracy necessary (NASA, 1995: sec.7, p.3), and with this knowledge ground-based computer processing can correct for discrepancies in received signals (between spacecraft). A constellation of this type would have the potential of creating an extremely large synthesized aperture; however, the processing power required for a constellation of more than two or three satellites would no doubt be tremendous and possibly prohibitive with current capabilities.

Further analysis may be performed regarding the types of orbits employed for tactical satellites. The primary orbit chosen as the baseline for this study was a Sun-synchronous orbit at an altitude of 350 kilometers. This orbit type has definite advantages for remote sensing and Earth resources missions; however, other orbits exist which may provide greater utility for specific missions. The highly eccentric Molniya orbits provide long dwell time, but at a geosynchronous altitude during that dwell time; this orbit type may be useful for tactically-applied communications satellites designed for short mission durations. "Walker" orbit parameters (Wertz, 1992: 189-192) provide a useful method for construction of constellations with specific coverage goals in mind. One example is that of multiple satellites being "staggered" in one or more specific orbital planes such that, as one satellite "sets " on a target, the next satellite in line comes into view. Constellation design and optimization is a complex art. Space planners and designers may wish to examine the advantages and disadvantages of the coverages, dwell times, revisit times, and environmental stresses associated with various orbit types.

21.3 Logistics and Operations

There are logistical challenges involved with the transport, storage, and maintenance of a (large) supply of not only tactical satellite buses, but also mission modules (of multiple types), small launch vehicles, and support equipment (and support aircraft in the case of air-launched spacelift). Another possible (albeit novel) logistical and/or operational strategy study would be to investigate possible ways of recovering the satellites -- either by retrieval or reentry -- and gauge their feasibility and utility in a tactical environment constrained by costs and the increasing desire to reuse hardware.

Further analysis may be performed on the actual missions and/or applications for the tactical satellite, not only on the sensing and support (of terrestrial forces) aspect of space based platforms, but also on the possible force application roles of such platforms.

The air-launched ICBM test program of the 1970s and the current "ALT-Air" program sponsored by the Ballistic Missile Defense Organization (BMDO) have demonstrated the feasibility of a carrier aircraft-based, palletized launch scheme. In this scheme, a launch vehicle is transported inside the carrier aircraft, deployed off of an aft-ejected, parachute-assisted pallet with a stabilized drogue chute, and ignited. This launch scheme could be applied to a new launch system specifically designed for this purpose. Investigations could include adaptation of an existing system to this scheme (like the ICBM tests of twenty years ago). This transport environment has the potential of providing a much more benign environment (vibrational and thermal) for the launcher and spacecraft payload, as well as possibly supporting a "clean" environment (e.g., a "clean tent" such as used by the *Pegasus*) inside the aircraft itself during transport -- due to the fact that the interior of the aircraft is environmentally controllable. Additionally, regular loading crews would need no special training for handling the rocket pallet, since the pallet would be the same as any other pallet fitted for that particular aircraft.

21.4 Mission Modules

This design study focuses specific design determination efforts upon only the satellite bus; however, due to the particular design paradigm, the primary design standards imposed upon the mission module designer may be enumerated. Though the standard small tactical spacecraft bus will provide support for the mission modules, mission module designers will be required to design "to the bus", as opposed to the bus being built for the specific payload equipment. Analogous to the interface requirements to which underwing stores on a fighter must be designed, the mission modules must conform to certain electrical, mechanical, data protocol, telemetric, and software interfaces incorporated into the bus design. Many of the specifications for these interfaces will be provided by the SPIG (see Tradeoffs).

The candidate bus designs were judged upon the amount of launch volume and launch mass which they reserved within the *Pegasus* payload fairing for the mission module. The implemented satellite bus design will provide both a launch volume "envelope" and a launch mass budget within both of which the volume and mass of the mission module must remain. Another primary performance design point for the mission module designer will be the amount of power available for proper operation of the mission module.

The mission module designers and builders will be expected to provide any software and/or satellite operating system patches or extensions specific to the mission being supported by a particular mission module. A software extension may, for example, include an update to the spacecraft's attitude control logic, specifying attitude tolerances (pointing accuracy, attitude knowledge, etc.). Depending on the specific mission for which the spacecraft and mission module will be employed, several different software extensions may be applicable within a single type of mission module (i.e., the software provided with a mission module will be expected to tailor the mission module to a specific mission). This requirement should provide space planners ample flexibility in the application of a single type of mission module. This flexibility will also provide planners with the capability to update mission parameters as theatrical situation and requirements change.

Thermal protection and radiation protection must also be incorporated into mission module design; the bus does not provide protection to the mission module from these two elements. These protective measures may include blankets, coatings, radiation hardening, cryogenics, etc. The methods employed for a given mission module will vary depending on the type of equipment employed, and will be left completely to the mission module designers.

One final capability which all mission module types must incorporate is the capability for a "quick startup." The tactical space planner must have the means to place a mission module into a desired position and begin delivery of mission data as soon as possible. This is a very nonspecific requirement, due to the different functional complexities associated with different types of mission modules, and one which will probably be determinable only after on-orbit experience has been gained with the new systems. Certainly the deployment of solar arrays, calibration of sensing equipment, calibration of attitude sensors, and initial power system loading procedures will preclude a mission module from performing its mission immediately after launch. But robust calibration and testing by integration crews before launch (or similar "streamlining" of initial checkout procedures), in conjunction with experience, may be able to minimize the commanding and testing time necessary to fully prepare the satellite. In this respect, minimizing this delay for initial satellite checkout is a design goal, not only for the mission modules but also for the satellite bus. This design goal is captured within the value system design by the objective "minimize preparation time to launch"; all activities leading up to full operational capability are considered to be the "launch" phase. Since this is an objective to be addressed in the design of the bus, the mission module designers should also address this objective within their designs.

Table 21-1 summarizes the generally specified requirements for the design of the various mission modules.

Table 21-1: Mission Module Design Requirements

Design Consideration	Mission Module Design Requirement
Mission Scope	single-sensor type; narrow mission specification
Mission Mass	under 120 kg
Mission Power	average under 320 W peak under 820 W
Mission Volume	under 0.6 m
High Data Rate Downlink	must be integral if greater than SGLS rate is desired
Data Storage Capacity	must be integral if greater than 2Gbytes is desired (unless storage is modular)
Mechanical Interface	conform to SPIG
Electrical Interface	28 V regulated bus standard; conform to SPIG
Telemetry/Software Interface	compatibility with bus standard formatting; specialized mission software extensions (to SOS) must be integral
Thermal Environment	isolation from bus; specialized mission equipment integral
Design Focus	tactical; minimize testing time; minimize warmup time

The mission modules modeled in the Modsat computer model are necessarily "generic" in nature to provide both flexibility in design evaluation and a foundation on which more specific types may be modeled in both the current version and in future versions.

Future operational analysis may focus on optimizing the functionality of different types of mission modules and putting together the most tactically useful, easily storable, quickly integratable, and technically feasible combination of mission modules for tactical space missions.

A specific mission module for performance of a "LASER designator from space" role was not specifically addressed by the system design study, due to the number and variability of the many factors involved in the mission analysis for such a mission module, as well as the "experimental" nature of any such mission module if constructed with current technology. A mission module of this type may be roughly modeled, however, with the LASER/LIDAR mission module tools incorporated in the Modsat computer model. A future trade study on the design of such a mission module would require analysis of 1) illumination efficiency, power, and wavelength(s); 2) target reflectivity/signature in the given wavelength(s); 3) detector positioning (azimuth, elevation, and altitude), sensitivity in the given wavelength(s), field of view, signal to noise ratio, and velocity; and 4) possible adversarial countermeasures and spoofing. Finally, sizing of the mission module's volume, mass, and power requirements may or may not make this application feasible for a small satellite application. Of course, a primary goal of this type of research would be the determination of "payoffs" in costs, manpower, equipment, capability, and responsiveness that this type of system may or may not achieve.

A similarly experimental application (and as worthy or more worthy of further investigation) being developed for LASERs is that of extremely high data rate communications systems. Many of the tradeoff factors and design considerations involved in the design of a LASER designation system are applicable to the design of a LASER-based communications system (i.e., power, sensitivity, positioning, signal to noise ratio, field of view, and , of course, atmospheric attenuation).

21.5 Small Satellite Technologies "On the Horizon"

Many novel and exciting (as well as very technically challenging) technologies promise to change the face of satellite design in the not so distant future. All of these technologies are expected to be developed within the next ten years.

Flywheel Technology -- Flywheels provide power and momentum storage through the utilization of kinetic energy storage. These structures represent potentially lighter weight and higher capacity than chemical-based batteries, with the added functionality of naturally stabilizing a spacecraft in (as do traditional momentum wheels). This "functional density" (in which one component performs more than one function) is a popular theme for small satellite design, and is already evident in most designs for spacecraft CPUs, as multifunctional microprocessors are becoming the norm for small satellites (Hively, 1996).

Lithium-Ion Batteries -- These batteries provide vastly higher capacity than traditional and current technology chemical batteries (see Vol II, Tradeoffs, Electrical Power Subsystem).

Inflatable Structures -- This technology will allow smaller spacecraft buses to support much larger, more capable active sensors, such as those required for synthetic aperture

RADAR. Current efforts are underway at NASA to produce electronically steerable, high-resolution RADARs for launch on small vehicles, but inflatable structure technology would significantly reduce payload mass and required payload fairing volume (scaling down required volume from "cubic feet" to volumes on the order of "cubic inches"), thereby freeing up space on the booster for other experiments (on the spacecraft) or other vehicles (within the fairing) (NASA, 1995).

Global Positioning System (GPS) Applications -- GPS promises to provide much better accuracy and more timely and autonomous orbital position prediction and tracking than current methods of ground tracking. Utilization of GPS will free up much of the overtaxed Air Force Satellite Control Network (AFSCN) from mundane "tracking" supports for the new vehicles equipped with GPS receivers. Experiments in the future will also include single vehicles equipped with multiple receivers, testing GPS capability to determine spacecraft attitude. If this application proves functional, it will relieve much of the attitude control system requirements for attitude knowledge sensors, thereby further reducing spacecraft mass.

"Toroidal" Propellant Tank -- This propellant tank design was borne out of system synthesis efforts as a theoretically more efficient propellant tank packaging scheme, optimizing available volume within a spacecraft.

Fourier Transform Hyperspectral Imaging -- This imaging package under investigation at Phillips Lab represents a new paradigm in multispectral imaging -- spectral resolution approaching that of gas chromatography and/or spectrometers (i.e., evolution toward a "continuous spectral imaging system" paradigm) through the usage of Fourier Transform systems for spectral separation. Improved spectral resolution, lighter instrument weight, and more efficient transmission (than the current "best" method of dispersion gratings) is achieved through Fourier Transform separation (Hagan, 1996;Otten and others, 1995).

Pulsed plasma thrusters -- These and other low-thrust, high specific impulse, non chemical propulsion systems will provide lower thrust, but more total delta-velocity capability (per unit mass) over the life of the satellite than chemical thrusters (Hagan, 1996). Experiments utilizing PPTs are planned for the Phillips Lab's MightySat program.

Ka-Band Transmitter Experiment -- Another experimental payload project for the Phillips Lab MightySat program, this phased array communications package will provide testing and validation of technologies expected to reduce the mass, moving parts, and spacecraft attitude adjustments required to track a signal from a communications ground station. It will also study high data rate modulation techniques.

Small liquid-fueled booster -- Solid rocket motors (SRMs), while inexpensive and easily adaptable to small launch systems, are heavy, toxic, fragile, less flexible, and less reliable (in general) than liquid rocket-based launch systems. Phillips Lab and other research organizations (as well as some sectors in industry) are developing prototypes of a simplified "blow down" pressurized liquid-fueled rocket for use as a small launch system (Warner, 1996; Worden, 1996). This system incorporates propellant, oxidizer, and an inert pressurant (helium) to inject the propellant and oxidizer into the combustion chamber. This system can be built with few or no moving parts, and, by virtue of being "throttleable" the system's performance can be fine-tuned and/or trimmed during flight (as opposed to a SRM, which may be vectorable, but not dynamically thrust-variable while in-flight), thereby increasing initial orbit accuracy. These types of simple, small rockets could eventually replace SRMs in most applications (e.g., as an air-launched system).

22. Conclusions

Although the individual products and ideas generated throughout this study may be individually worthy of merit, the three important results of this study fully characterize the synergism of the effort. The utilization of an adaptable (i.e., specific to the task at hand and the circumstances of the environment) System Design Process made all of the effort possible and productive. The generation of a feasible, value-added, "clean-sheet" MIDTAC design for a small tactical satellite bus provides a basis for further development of "tactical space" or "TacSpace" concepts. The construction of a generic and modular (i.e., robust, modifiable, expandable) Modsat Computer Design Model, though the greatest challenge of the effort, provides a useful, valuable design platform for use by both future researchers and students.

22.1 Modsat Model

The construction effort involved with the development of a fully integrated computer design and analysis tool for small satellites comprised a systematic design process in itself. As with the individual subsystem component choices, individual subsystem modeling sections formed along baseline component characteristics determined by the subsystem trade studies. The value system determined by the team formed the basis of the analysis section of the model. The Modsat modeling software package provides for analysis of physical characteristics, mission performance, and overall costs.

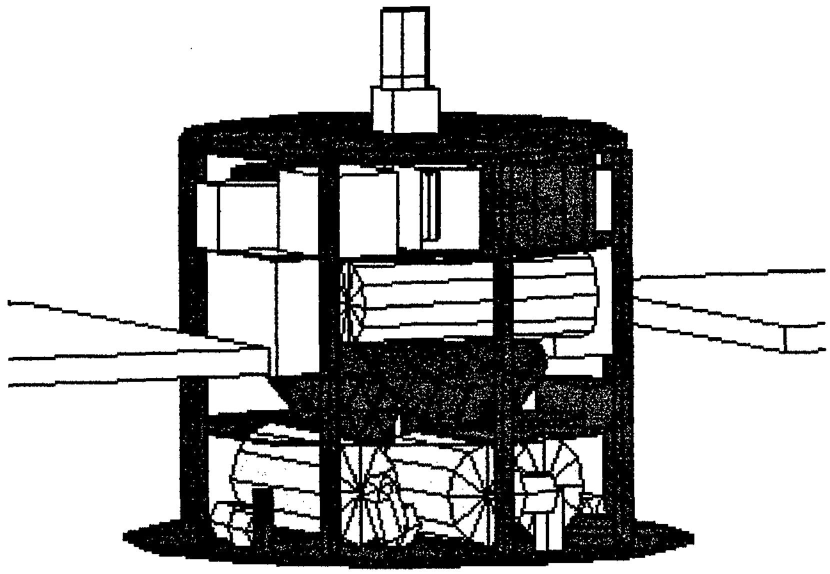
The Modsat model provides a foundation for further analysis efforts. The underlying functions of the software may be modified or expanded according to a particular user's requirements and objectives. A vast array of different sizes, shapes, materials, and other characteristics may be modeled, based upon user input. The initial version of the model provides estimations which may be updated (through modification of the underlying code) with evolutions in space technology or changes in design philosophy. Aside from these more esoteric considerations, the software, of course, may be used for the analysis of small satellite designs other than those analyzed in this preliminary study. Using differently modeled components and differently modeled value systems, the Modsat modeling tool has the capability to produce a wide range of possible designs.

22.2 Design Concept

The MIDTAC spacecraft bus provides a generic design that may be further developed for specific applications or more completely engineered for actual production. The MIDTAC bus, as a complete system, has been designed to the extent that feasible bus enhancements may be easily explored and developed. As recommended by this system study, the expected implementation of the MIDTAC bus incorporates a modular power generation system to provide tailoring of power levels required by various mission modules. The actual design may also incorporate other modular and/or tailorable components or subsystems.

The next step in the development of this bus design should be an engineering study of the interfaces required to integrate all of the individual satellite subsystems and components. While the MIDTAC design includes physical characteristics and a "working concept", this design is only the first step in development process; it comprises the results of an initial "concept exploration" phase for a new space system. Concurrent engineering studies concerning manufacturing, integration, and testing for the MIDTAC design must also be accomplished. Finally, mission modules must be designed for flight testing and operation on the MIDTAC bus.

Figure 22-1: MIDTAC Bus and Vital Statistics



Physical Characteristics/Capabilities:
Mass: 243 kg
Available Power (average/peak): 319/827 W (tailorable to mission)
Available Mission Mass: 75 kg (350 km, sun-synch)/200 kg (350 km, 28.5 deg)
Available Mission Volume: 0.7855 m^3
Pointing Accuracy/Attitude Knowledge: 0.1 deg/0.05 deg (nominal)
Data Storage: Modular 2Gbyte SSDR (tailorable to mission)
Delta-velocity Capacity: 300m/s

Subsystem Features:
Mission Modules: SPIG interfaces, EO, MSI, LASER/LIDAR, SAR
ADCS: three-axis stabilized, reaction wheels (4), star sensor, Earth sensor, IMU
Propulsion: monopropellant thrusters (6), cylindrical tanks (4)
Structural: octagonal "cage" structure, "3-level" shelf system, SMASH option
EPS: NiH_2 batteries, modular GaAs solar arrays, decentralized distribution
TT&C: SGLS compatible, 1024 kbps nominal downlink, Satellite Operating System (SOS), TCP/IP protocol

The MIDTAC design ultimately provides a solution to one portion of a much greater, underlying problem facing modern space efforts (military, scientific, and commercial). More responsive, less expensive, and more efficient access to space is an issue which requires new and innovative approaches to not only spacecraft design (the focus of this study), but also spacelift, command and control, and information processing and distribution. In addition to providing a tactically applicable satellite bus, the MIDTAC design will provide the Air Force with ready-to-fly space research platforms. The space environment will be more accessible to technology demonstrations, developmental payloads, and other space experiments by having these standard buses (with standard mission interfaces) readily available. The MIDTAC design essentially provides to the Air Force a standard vehicle -- putting the "horse" before the "cart" -- on which it may more readily and effectively conduct space operations and technology development. MIDTAC will allow the Air Force to more quickly accumulate valuable "spaceflight time" and experience. Only with increased "hands on" experience in spaceflight and space operations will the Air Force fully evolve into its role as a "Spacepower". MIDTAC provides the means to that end.

22.3 System Process and Beyond

The start of the system design began necessarily with a generalized, high-level treatment of the problem posed by the CDM: the generation of a "clean sheet" tactical satellite design for use as a "multirole" satellite, capable of supporting a wide variety of payload (mission module) types. This design should be easily and inexpensively produced for the Air Force, smoothly integrated by a primarily "blue suit" special weapons crew, and quickly launched for either quick-response or asset-replenishment missions. Initial considerations involved the various high-level (nonspecific components) design and implementation approaches to this problem; further efforts evolved to focus efforts on creating the "baseline" or "point design" with which the original design approaches could be considered in combination. This combination of design (the MIDTAC) and approach (i.e., modularized components) produced an optimal solution.

The (satellite) functional division of effort, in addition to the assignment of system process responsibilities among team members, was key to success. The convergence of the overall design and the development of the Modsat modeling software required expert knowledge of individual subsystems. Consideration of generic remote sensing mission modules provided baseline requirements for the bus design. The system, reliability, and subsystem trade studies narrowed the solution space for the bus design to a point where baseline designs could be generated. Effective coordination of the overall system design effort required central coordination for each of the system process phases: problem definition, value system design, system and subsystem design tradeoffs, system synthesis, modeling and optimization, decision making, and implementation. This approach not only allowed the systematic construction of a robust design and analysis tool, but also the synthesis of seven fully integrated, fully characterized designs which could then be thoroughly analyzed and evaluated.

The basic validity and robustness of the process may be fully and objectively realized when considering the role of the CDM. Other than an initial, broad design philosophy, the CDM provided little input. Faced with this "ill-posed" problem, the team created a process which allowed a "natural" evolution of objectives, focusing of efforts, convergence of designs, and, most importantly, solidification of goals. These goals included the construction of the Modsat model and the development of a "baseline" design. Modification of either the assumptions of the problem definition or the value system (based upon the preferences of the CDM) may yield vastly different results. This allows the adaptability of the process to other space system design projects.

This process is unique, innovative, and goes beyond traditional system and satellite design approaches. Although Hall's Process and the SMAD process were used as references and process "baselines", these approaches ultimately proved unsuitable, due to the unusual design paradigm assumed by the team. The "mission module" is a concept wherein the payload equipment must be integrated to the satellite bus and conform to bus-constrained design parameters. This is a design paradigm unpopular within the aerospace industry, and runs contrary to the traditional approaches to spacecraft design prescribed by the SMAD process. Considering the stagnation of space efforts and the lack of relief from the high cost and low availability of space access, the pursuit of an alternative spacecraft design paradigm was reasonable, if not necessary. The team required, and ultimately developed, a process that was less dependent upon mission-derived design specifications and allowed more design freedom to consider the wide range of spacecraft design possibilities.

Finally, the process is, unlike the SMAD process, synergistic in approach, and it epitomizes the concept of "functional density," wherein the process served many purposes simultaneously. The research involved in the subsystem trade studies produced the component data for use in the modeling tool, narrowed the scope of subsystem design choices, and refined the design objectives and value system. The iterative process of the synthesis of individual designs solidified design alternatives and also refined the model. The ultimate results of the development and application of this process resonate beyond the generation of the MIDTAC design and the development of the Modsat model. The results of this study show that, as a solution to the "space access" problem, an alternative to the current design paradigm (i.e., "generic" versus "specific") is not only feasible but desirable. This study, its process, and its products set the standards of creativity, innovativeness, adaptability, and robustness for future spacecraft design efforts. The Modsat design team has set the bold example that others will follow.

APPENDIX A: CRITERIA PREFERENCE IN THE DESIGN OF A SMALL, STANDARDIZED, TACTICAL SATELLITE BUS

We are conducting a systems study to create a preliminary design concept for a small, starndardized, tactical satellite bus. You could greatly assist us by filling out the Preference Charts on the following pages. By doing so you will give us an idea of the relative priorities you place on the criteria we have chosen to evaluate our solutions. The example below explains how to fill out a preference chart. Your experience and opinions are highly valued.

HOW TO FILL OUT A PREFERENCE CHART

Fill in each cell with one of the following symbols:

-- means *"much less important"*

- means *"less important"*

= means *"equally important"*

+ means *"more important"*

++ means *"much more important"*

For example, a "+" in cell "ij" means that the criteria in row i *more important* that than the criteria in column j

Note that you only have to fill in cells above or below the diagonal. The cells on the other side of the diagonal are just the inverse of the cells you filled in.

EXAMPLE

Suppose that you established 4 criteria for buying a car (1) minimize cost, (2) maximize gas mileage, (3) Appearance, (4) Towing capacity.

Buying a Car

	Min Cost	Max Gas Mileage	Appearance	Towing Capacity
Min Cost		+	++	+
Max Gas Mileage	-		+	=
Appearance	- -	-		-
Towing Capacity	-	=	+	

This table states that the individual who filled out this chart feels:

Minimizing Cost is <u>more important</u> than **Maximizing Gas Mileage**
Minimizing Cost is <u>much more important</u> than **Appearance**
Minimizing Cost is <u>more important</u> than **Towing Capacity**
Maximizing Gas mileage is <u>more important</u> that **Appearance**
Maximizing Gas mileage is <u>equally important</u> as **Towing Capacity**
Appearance is <u>less important</u> than **Towing Capacity**

CRITERIA DESCRIPTIONS

The main objective, which encompasses all lower level objectives, is:
" DEVELOP THE BEST STANDARDIZED BUS FOR SMALL TACTICAL SATELLITES ".

The key words are defined as follows:
BUS: The bus is that collection of satellite components whose functions are generally common to all satellites. The bus provides all the housekeeping and support functions necessary to operate the mission payload, hereafter referred to as the mission module.
STANDARDIZED: The bus will be a common platform with a standardized interface, to which many different types of mission modules will be integrated.
THE BEST: System solutions must focus on satisfying the requirements and objectives of the decision maker, which are themselves subject to refinement and qualification.
SMALL: Satellites employing this bus must be in the weight class of Pegasus or Lockheed Martin launch vehicles.
TACTICAL: Satellites employing this bus will be used for tactical misisons.

The fundamental objectives are shown below in Figure 1.

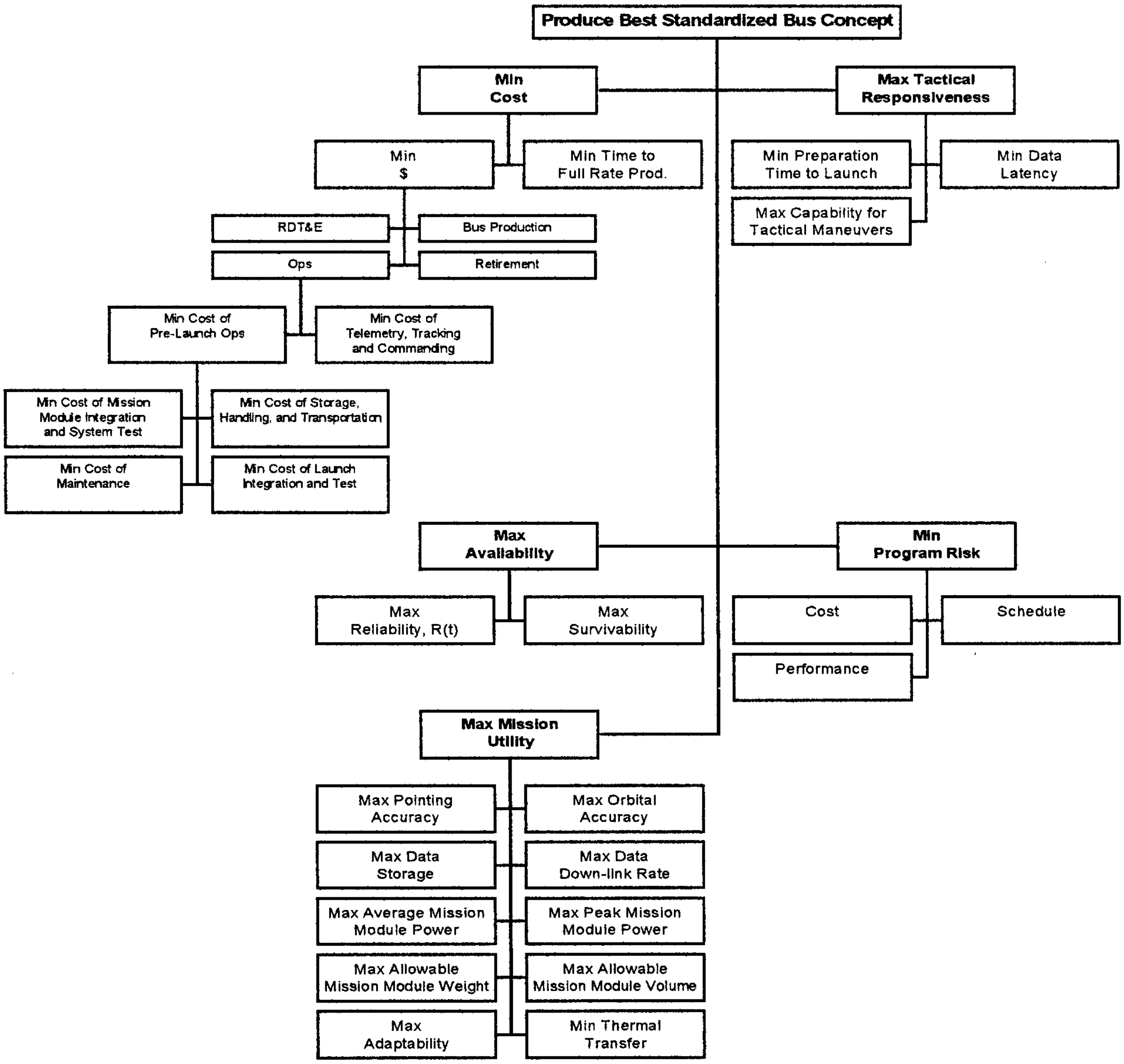


Figure 1: Overall objective hierarchy

Objectives that occupy the same level of the overall objective hierarchy will be compared to each other.

Minimize Cost:

Cost is broken down into monetary value and time.

1. Minimize $: The performance of each monetary cost objective may not be measured in actual dollars; for some objectives a proxy utility scale will describe the cost. The main subobjectives are:

a. Min RDT&E Cost: Includes all non-recurring costs from the beginning of the project to the production effort.

b. Min Bus Production Cost: The cost of manufacturing the final product.

c. Min Retirement Cost: At the end of its mission and/or design life, the satellite will be retired via reentry into the atmosphere, natural orbital decay, or placing it in a "retirement" orbit. The method of retirement will dictate the cost.

d. Operations Cost: This cost can be divided into pre- and post-launch costs.

	MINIMIZE R D T & E COST	MINIMIZE BUS PRODUCTION COST	MINIMIZE RETIREMENT COST	MINIMIZE OPERATIONS COST
MINIMIZE R D T & E COST				
MINIMIZE BUS PRODUCTION COST				
MINIMIZE RETIREMENT COST				
MINIMIZE OPERATIONS COST				

(1) Minimize Pre-Launch Operations Cost: This is the sum of all ground expenditures necessary to prepare the satellite for launch. The subobjectives for this cost are:

(a) Minimize Cost of Mission Module Integration and System Test: This cost covers all efforts to mate the mission module to the bus, and to test the integrated satellite. There are various manpower, equipment, software, and overhead costs associated with connecting the power, signal, thermal and structural interfaces. The testing effort includes all actions to ensure the integrated satellite is ready to be shipped for launch.

(b) Minimize Cost of Storage, Handling, and Transportation: Includes facilities, equipment, special procedures, manpower and overhead.

(c) Minimize Cost of Maintenance : There are various costs associated with maintaining MODSAT hardware and software, including manpower, spares, and supplies. This covers both periodic maintenance and repairs.

(d) Minimize Cost of Launch Integration and Test : This covers all efforts to mate the satellite to the launch vehicle, and to test the loaded rocket.

	MINIMIZE COST MIS.MOD. INTEG&TEST	MINIMIZE COST STORG&HAND&TRANS	MINIMIZE COST MAINTENANCE	MINIMIZE COST LAUNCH INTEG.&TEST
MINIMIZE COST MIS.MOD. INTEG&TEST				
MINIMIZE COST STORG&HAND&TRANS				
MINIMIZE COST MAINTENANCE				
MINIMIZE COST LAUNCH INTEG.&TEST				

(2) Minimize Cost of Telemetry, Tracking, and Commanding: This cost covers all operational expenditures for tracking, commanding, and maintaining the orbiting satellite. Elements of TT&C include ground segment hardware and software, satellite operations personnel, and communications and data flow operations.

	MINIMIZE PRELAUNCH OPERATION COST	MINIMIZE COST T T & C
MINIMIZE PRELAUNCH OPERATION COST		
MINIMIZE COST T T & C		

Minimize Time to Full Rate Production: The time required to achieve full rate production of the bus should be minimized.

	MINIMIZE $	MIN. TIME TO FULL RATE PRODUCTION
MINIMIZE $		
MIN. TIME TO FULL RATE PRODUCTION		

Maximize Tactical Responsiveness

The bus is intended for tactical applications. Thus, responsiveness is a primary objective. Satellites using this bus must be able to respond quickly to rapidly generated needs and mission requirements.

1. Minimize Preparation Time to Launch: This time interval begins with the demand for a particular mission, and ends with the delivery of the satellite for launch vehicle integration.

2. Minimize Data Latency: Data latency refers to the time between mission data collection and reception by the user in unusable form. This objective intends to capture the differences in data processing, down-link, and delivery architecture.

Maximize Capability For Tactical Maneuvers: The satellite may be called on to perform changes in station / inclination in response to tactical mission needs. This subobjective covers the satellite's slew capability in response to different targeting needs.

	MINIMIZE PREPARATION TIME TO LAUNCH	MINIMIZE DATA LATENCY	MAXIMIZE CAPABILITY FOR TAC. MANEUVERS
MINIMIZE PREPARATION TIME TO LAUNCH			
MINIMIZE DATA LATENCY			
MAXIMIZE CAPABILITY FOR TAC. MANEUVERS			

Maximize Availability

A sound system design will attempt to maximize on-orbit availability.

1. Max Reliability: For the purposes of this study, reliability refers to the probability that, given a non-hostile environment, the bus will be able to perform its primary mission of supporting the mission module at a given point in its lifetime. It is a measure of the hardness of the bus to the natural space environment over the satellite's lifetime, within a specified confidence level

2. Maximize Survivability: Survivability refers to the ability of the system to perform its intended mission after exposure to stressing environments created by enemy or hostile agent.

	MAXIMIZE RELIABILITY	MAXIMIZE SURVIVABILITY
MAXIMIZE RELIABILITY		
MAXIMIZE SURVIVABILITY		

Minimize Program Risk

Program risk refers to potential for elements of the program to fail to come together as planned. Risk can be assessed in the areas of cost, schedule, and performance.

1. Cost Risk: Each system development program has cost risk, in that the predicted life cycle costs or individual elements of the cost may be much higher than planned. In a sense, cost risk is a measure of the lack of confidence in the cost estimates.

2. Schedule Risk: Schedule risk refers to the potential for expensive, unforeseen slips in the development schedule.

3. Performance Risk: There is a chance that some technological aspect of the system, be it hardware or software, will not work as well as planned. This usually depends on the maturity of the technology in question.

	COST RISK	SCHEDULE RISK	PERFORMACE RISK
COST RISK			
SCHEDULE RISK			
PERFORMACE RISK			

Maximize Mission Utility

Mission utility refers to the ability of the bus to accommodate a range of different missions modules. In other words, it is a way of quantifying how well the bus performs its role of being generic and standard. The larger the range of possible missions, the higher the mission utility. Mission utility is supported by the various aspects of bus performance, such as pointing accuracy and available power. In other words, if more performance capability is built into the bus, more mission types can be accommodated. Thus, several performance subobjectives were adopted, in addition to the obvious desire to maximize the available weight and volume for the mission module.

1. Max Pointing Accuracy: Many space mission applications require a great deal of spacecraft pointing accuracy, for precise pointing and orientation of sensing instruments.

2. Max Orbital Accuracy: The satellite must be able to maintain its intended orbit, in order to be available for its intended applications.

3. Max Data Storage: Various missions may require the spacecraft processor to temporarily store mission data, for later transmission.

4. Max Data Down-link Rate: Although the bus must be compatible with the Air Force Satellite Control Network (AFSCN), some missions may require a higher data downlink rate than can be accommodated with AFSCN compatible communications equipment. Systems solutions that provide for higher rates would be more attractive, all other factors being equal.

5. Max Average Mission Module Power: This objective strives to maximize the amount of power, on average, that the bus makes available to the mission module. A survey of existing LEO military satellites reveals that their power requirements cover a wide range.

6. Max Peak Mission Module Power: Some mission applications will require large amounts of power during peak operations. The bus must be able to handle peak demands which can greatly exceed the average demand.

7. Max Allowable Mission Module Weight: One of the main design goals of this study is a lightweight bus which allows for the heaviest possible mission module. The amount of allowable mission module weight is determined by subtracting the bus weight from the total allowable weight, which depends on the launch vehicle and the selected orbit.

8. Max Allowable Mission Module Volume: This subobjective is met by minimizing the volume of the bus. The amount of allowable mission module volume is determined by subtracting the bus volume from the total allowable volume, which depends on the volume and dimensions of the launch vehicle payload fairing.

9. Max Adaptability: This subobjective is intended to capture the desire for flexibility of design, and adaptability to changing mission requirements. The ability to add needed capability or remove excess capability is valued, but consideration must be given to minimizing the amount of engineering which must be applied to the integration effort.

10. Min Thermal Transfer: It is necessary to limit the amount of heat energy which crosses the interface to the mission module. Mission instruments can be very sensitive to

heat. Consideration must be given to the highly dynamic nature of a spacecraft thermal environment.

	POINTING ACCURACY	ORBITAL ACCURACY	DATA STORAGE	DATA DOWN-LINK RATE	AVERAGE MIS. MOD.POWER	PEAK MIS.MOD. POWER	ALLOWABL. MIS. MOD. WEIGHT	ALLOWABL. MIS. MOD.VOLUME	MAX ADAPTABILITY	MIN THERMAL TRANSFER
POINTING ACCURACY										
ORBITAL ACCURACY										
DATA STORAGE										
DATA DOWN-LINK RATE										
AVERAGE MIS. MOD. POWER										
PEAK MIS. MOD. POWER										
ALLOABLE MIS. MOD. WEIGHT										
ALLOABLE MIS. MOD. VOLUME										
MAX ADAPTABILITY										
MIN THERMAL TRANSFER										

Fundamental Objectives

	MINIMIZE COST	MAXIMIZE RESPONSIVENESS	MAXIMIZE AVAILABLITY	MINIMIZE PROGRAM RISK	MAXIMIZE MIS. UTILITY
MINIMIZE COST					
MAXIMIZE RESPONSIVENESS					
MAXIMIZE AVAILABLITY					
MINIMIZE PROGRAM RISK					
MAXIMIZE MIS. UTILITY					

Structure's Trade Study (Phase I)

1.1 Calculating maximum volume for the payload bay

$\text{payload\_bay\_ht} := 34.27\,\text{in}$ $\quad$ $\text{payload\_bay\_dia} := 39.5\,\text{in}$

$$\text{volume\_payload\_bay} := \left(\frac{\text{payload\_bay\_dia}}{2}\right)^2 \cdot \pi \cdot \text{payload\_bay\_ht}$$

$$\text{volume\_payload\_bay} = 6.88176 \cdot 10^5 \cdot \text{cm}^3$$

1.2 Calculating surface area for the payload bay

$$\text{area\_payload\_bay} := 2 \cdot \left[\pi \cdot \left(\frac{\text{payload\_bay\_dia}}{2}\right)^2\right] + \pi \cdot \text{payload\_bay\_dia} \cdot \text{payload\_bay\_ht}$$

$$\text{area\_payload\_bay} = 4.32483 \cdot 10^4 \cdot \text{cm}^2$$

2.1 Calculating maximum volume and area for a hexagon

$\text{num\_sides} := 6$ $\quad$ $\text{polygon\_rad} := \dfrac{\text{payload\_bay\_dia}}{2}$ $\quad$ $\text{central\_angle} := \dfrac{2 \cdot \pi}{\text{num\_sides}}$

$\text{central\_angle} = 60 \cdot \text{deg}$ $\quad$ $\text{alpha} := \dfrac{\text{central\_angle}}{2}$ $\quad$ $\text{alpha} = 30 \cdot \text{deg}$

$\text{polygon\_base} := 2 \cdot \text{polygon\_rad} \cdot \sin(\text{alpha})$ $\quad$ $\text{polygon\_base} = 50.165 \cdot \text{cm}$

$$\text{polygon\_cntr\_to\_base} := \left(\frac{\text{polygon\_base}}{2}\right) \cdot \cot(\text{alpha})$$

$$\text{polygon\_cntr\_to\_base} = 43.44416 \cdot \text{cm}$$

$$\text{polygon\_area\_bottom} := \frac{\text{polygon\_base}}{2} \cdot \text{polygon\_cntr\_to\_base} \cdot \text{num\_sides}$$

$$\text{polygon\_area\_bottom} = 6.53813 \cdot 10^3 \cdot \text{cm}^2$$

$$\text{volume\_polygon} := \text{polygon\_area\_bottom} \cdot \text{payload\_bay\_ht}$$

$$\text{volume\_polygon} = 5.69117 \cdot 10^5 \cdot \text{cm}^3$$

$\text{polygon\_perimeter} := \text{num\_sides} \cdot \text{polygon\_base}$ $\quad$ $\text{polygon\_perimeter} = 300.99 \cdot \text{cm}$

$$\text{area\_polygon} := \text{polygon\_perimeter} \cdot \text{payload\_bay\_ht} + 2 \cdot \text{polygon\_area\_bottom}$$

$area\_polygon = 3.92762 \cdot 10^{4} \cdot cm^{2}$

2.2 Total length of all edges of a Hexagon

$tot\_length := 2 \cdot polygon\_perimeter + num\_sides \cdot payload\_bay\_ht$

$tot\_length = 1.12425 \cdot 10^{3} \cdot cm$

3.1 Calculating maximum volume and area for the sphere

$sphere\_dia := payload\_bay\_ht$ $\qquad volume\_sphere := \frac{4 \cdot \pi}{3} \cdot \left[\left(\frac{sphere\_dia}{2} \right)^{3} \right]$

$volume\_sphere = 3.45336 \cdot 10^{5} \cdot cm^{3}$ $\qquad area\_sphere := \pi \cdot sphere\_dia^{2}$

$area\_sphere = 2.38038 \cdot 10^{4} \cdot cm^{2}$

3.2 Total length of all edges of a sphere

$num\_rings := 10$ $\qquad tot\_length\_sphere := num\_rings \cdot (\pi \cdot sphere\_dia)$

$tot\_length\_sphere = 2.73462 \cdot 10^{3} \cdot cm$

4.1 Calculating maximum volume and area for a rectangle

$num\_sides := 4$ $\qquad polygon\_rad := \frac{payload\_bay\_dia}{2}$ $\qquad central\_angle := \frac{2 \cdot \pi}{num\_sides}$

$central\_angle = 90 \cdot deg$ $\qquad alpha := \frac{central\_angle}{2}$ $\qquad alpha = 45 \cdot deg$

$polygon\_base := 2 \cdot polygon\_rad \cdot \sin(alpha)$ $\qquad polygon\_base = 70.94402 \cdot cm$

$polygon\_cntr\_to\_base := \left(\frac{polygon\_base}{2} \right) \cdot \cot(alpha)$

$polygon\_cntr\_to\_base = 35.47201 \cdot cm$

$polygon\_area\_bottom := \frac{num\_sides \cdot polygon\_base \cdot polygon\_cntr\_to\_base}{2}$

$polygon\_area\_bottom = 5.03305 \cdot 10^{3} \cdot cm^{2}$

$volume\_polygon := polygon\_area\_bottom \cdot payload\_bay\_ht$

$volume\_polygon = 4.38106 \cdot 10^{5} \cdot cm^{3}$

$polygon\_perimeter := num\_sides \cdot polygon\_base$

$polygon\_perimeter = 283.77609 \cdot cm$

$area\_polygon := polygon\_perimeter \cdot payload\_bay\_ht + 2 \cdot polygon\_area\_bottom$

$area\_polygon = 3.47676 \cdot 10^{4} \cdot cm^{2}$

4.2 Total length of all edges of a rectangle

$tot\_length\_rectangle = 2 \cdot polygon\_perimeter + num\_sides \cdot payload\_bay\_ht$

$tot\_length\_rectangle = 915.73539 \cdot cm$

5.1 Calculating volume and surface area for a cylinder

$cylinder\_dia := payload\_bay\_dia$ $\qquad volume\_cylinder := \left(\frac{cylinder\_dia}{2}\right)^2 \cdot \pi \cdot payload\_bay\_ht$

$volume\_cylinder = 6.88176 \cdot 10^5 \cdot cm^3$

$area\_cylinder := 2 \cdot \left[\pi \cdot \left(\frac{cylinder\_dia}{2}\right)^2\right] + \pi \cdot cylinder\_dia \cdot payload\_bay\_ht$

$area\_cylinder = 4.32483 \cdot 10^4 \cdot cm^2$

5.2 Total length of all edges of a cylinder

$num\_rings := 10$ $\qquad tot\_length\_cylinder = 2 \cdot \pi \cdot cylinder\_dia + num\_rings \cdot payload\_bay\_ht$

$tot\_length\_cylinder = 1.50085 \cdot 10^3 \cdot cm$

Structure's Pairwise Comparison

Modularity and subcomponent supportability

	Cage	Truss	Blocks	Shelf	Drawer	Total
Cage		3	3	4	3	13
Truss	1		1	1	1	4
Blocks	1	3		3	2	9
Shelf	0	3	1		1	5
Drawer	1	3	2	3		9

Material usage

	Cage	Truss	Blocks	Shelf	Drawer	Total
Cage		1	3	2	3	9
Truss	3		4	3	4	14
Blocks	1	0		1	1	3
Shelf	2	1	3		3	9
Drawer	1	0	3	1		5

Structural Rigidity

	Cage	Truss	Blocks	Shelf	Drawer	Total
Cage		1	0	2	2	5
Truss	3		1	3	3	10
Blocks	4	3		4	4	15
Shelf	2	1	0		2	3
Drawer	2	1	0	2		5

Manufacturing

	Cage	Truss	Blocks	Shelf	Drawer	Total
Cage		3	4	3	4	14
Truss	1		4	3	3	11
Blocks	0	0		1	2	3
Shelf	1	1	3		3	8
Drawer	0	1	2	1		4

Overall pairwise comparison results

	Modularity	**Materials Usage**	**Structural Rigidity**	**Manufacturing**	**Total**
Cage	**13**	**9**	**5**	**14**	**41**
Truss	**4**	**14**	**10**	**11**	**38**
Blocks	**9**	**3**	**15**	**3**	**30**
Shelf	**5**	**9**	**3**	**8**	**25**
Drawer	**9**	**5**	**5**	**4**	**23**

Obtaining the optimum beam diameter and thickness

$\text{gravity} := 9.81 \cdot \frac{\text{m}}{\text{sec}^2}$ factor_safety := 1.6 axial_g := 13

$E := 72 \cdot 10^9 \cdot \frac{\text{kg} \cdot \frac{\text{m}}{\text{sec}^2}}{\text{m}^2}$ $Pcr := 225 \cdot \text{kg} \cdot \text{gravity}(\text{factor\_safety} \cdot \text{axial\_g})$

$Pcr = 4.59108 \cdot 10^4 \cdot \text{kg} \cdot \text{m} \cdot \text{sec}^{-2}$ **<--Max loading conditions**

$i := 1 .. 150$ $\text{beam\_radius}_i := \frac{i}{100} \cdot \text{cm}$ **<--Beam radius 0.0 to 1.5 cm**

$j := 1..12$ $\quad$ $num\_sides_j := j$ $\quad$ **<-- 1 to 12 sides**

$L := 65 \cdot cm$ $\quad$ $beam\_load_j := \dfrac{Pcr}{num\_sides_j}$ $\quad$ **<--Load/beam**

$$I\_beam_j := \frac{beam\_load_j \cdot 4 \cdot L^2}{\pi^2 \cdot E}$$

I_beam: backing out this information so that beam thickness can be calculated for each I_beam case.

$I\_beam =$ (in $\cdot m^4$)

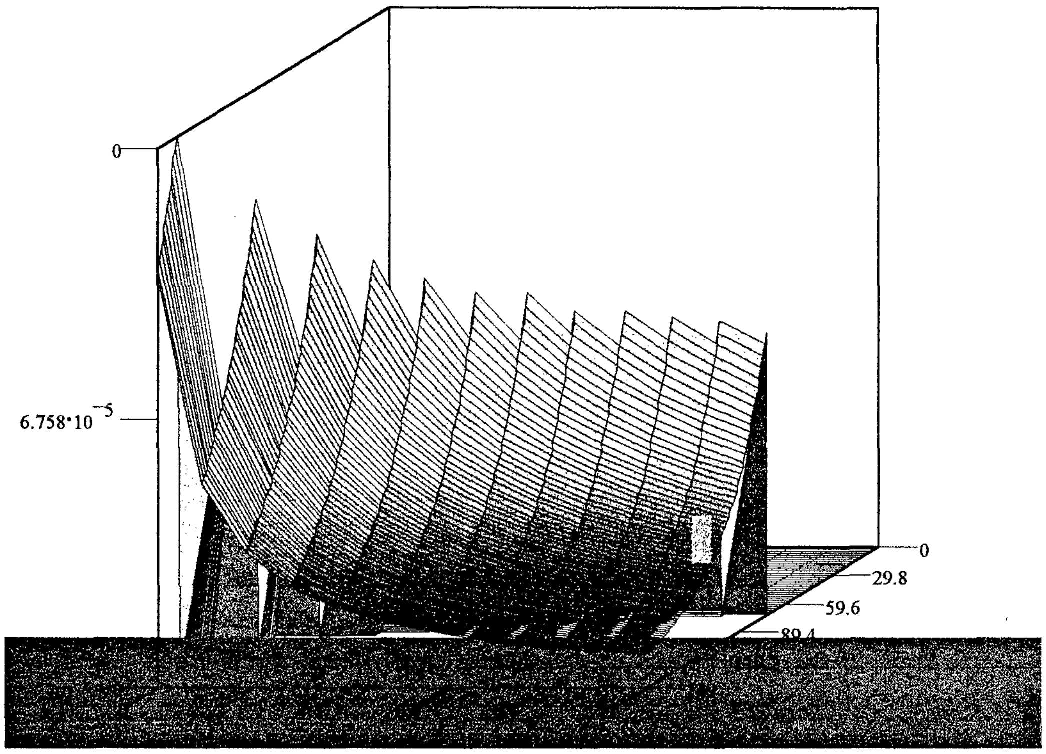
	1
4	$2.7296649 \cdot 10^{-8}$
5	$2.1837319 \cdot 10^{-8}$
6	$1.8197766 \cdot 10^{-8}$
7	$1.5598085 \cdot 10^{-8}$
8	$1.3648324 \cdot 10^{-8}$
9	$1.2131844 \cdot 10^{-8}$
10	$1.091866 \cdot 10^{-8}$
11	$9.9260541 \cdot 10^{-9}$
12	$9.0988829 \cdot 10^{-9}$

$$temp\_thick_{i,j} := \frac{I\_beam_j}{\pi \cdot (beam\_radius_i)^3}$$ **<--Find all possible thicknesses**

$$beam\_thick_{i,j} := \begin{cases} temp\_thick_{i,j} \cdot cm & \text{if } beam\_radius_i > temp\_thick_{i,j} \\ 0 \cdot cm & \text{if } temp\_thick_{i,j} > beam\_radius_i \end{cases}$$

Because the solved thicknesses may be greater than beam radius, the above relationship filters those conditions, setting them to 0.

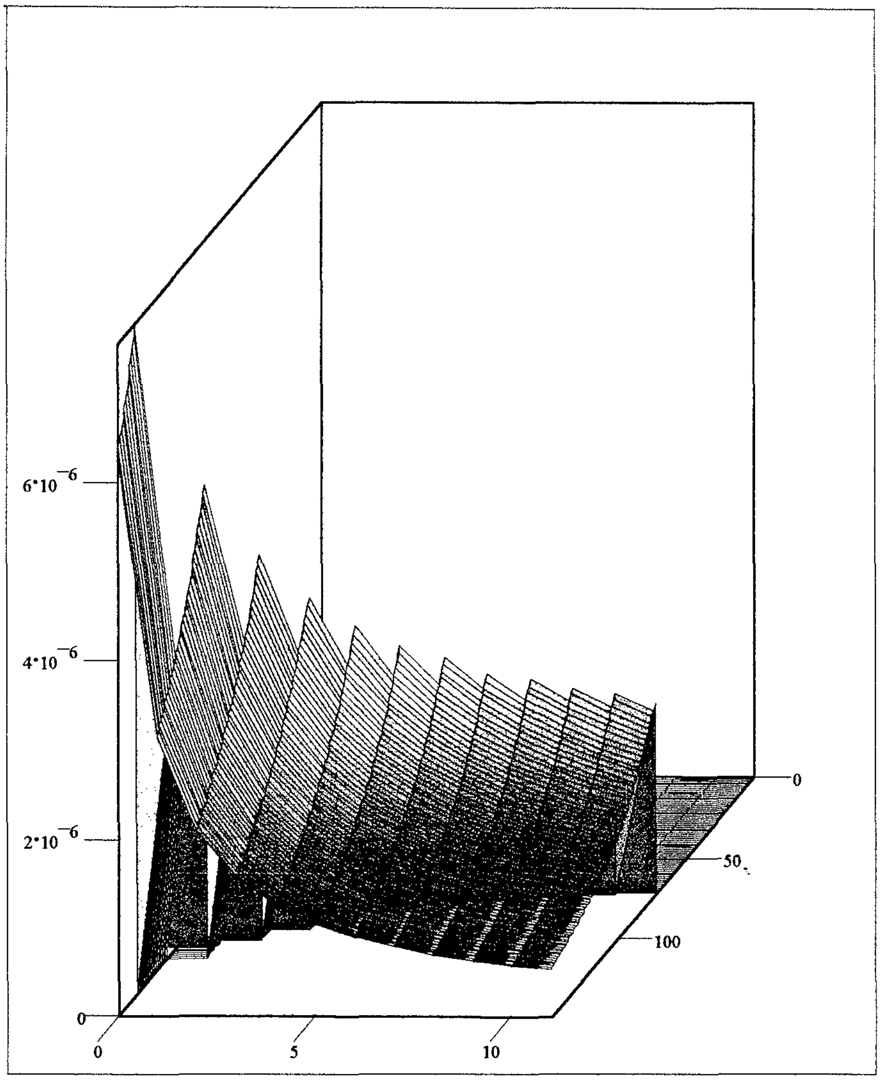
Beam thicknesses



beam_thick

$$temp\_radius_{i,j} := \frac{i}{100} \cdot cm$$

$$volume_{i,j} := \left[\left(temp\_radius_{i,j}\right)^2 \cdot \pi - \left(temp\_radius_{i,j} - beam\_thick_{i,j}\right)^2 \cdot \pi \right] \cdot L$$



volume

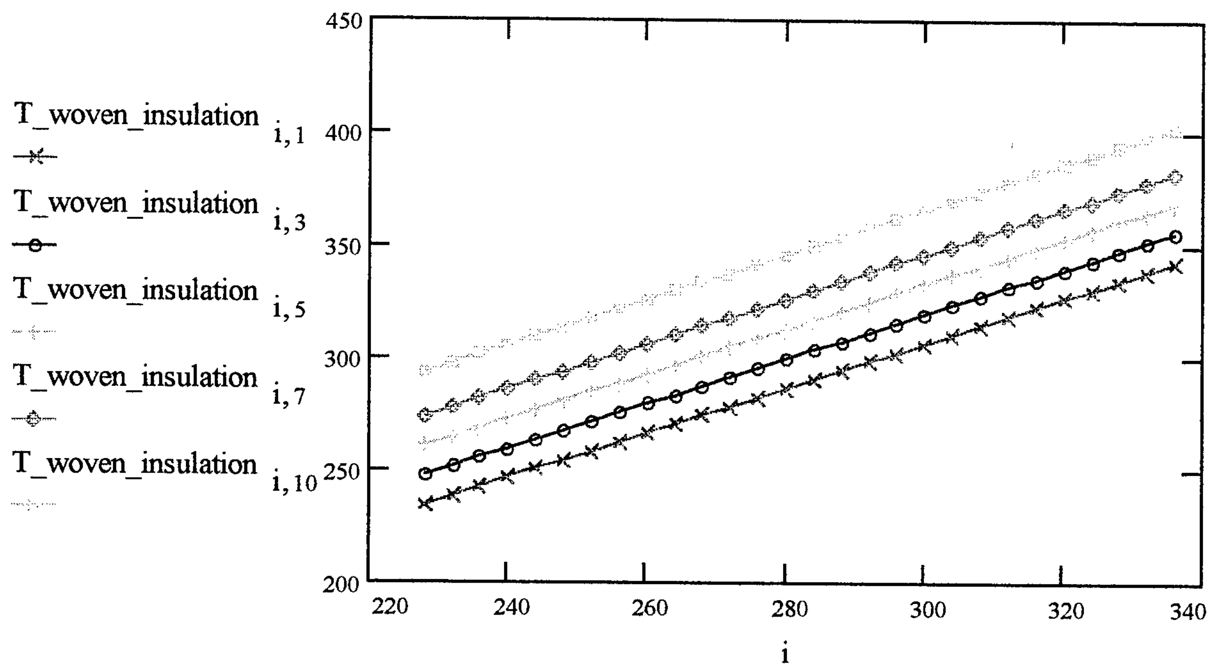
APPENDIX C: THERMAL ANALYSIS OF THE INTERFACE BLANKET

$Q\_dot := 15 \cdot watt$ $\quad k\_aluminum = 173 \cdot \dfrac{watt}{m \cdot K}$ $\quad k\_woven\_insulation = 0.038 \dfrac{watt}{m \cdot K}$

$i := 228, 232 .. 338$ $\quad T\_internal\_sat_i := i \cdot K$ $\quad j := 1 .. 10$

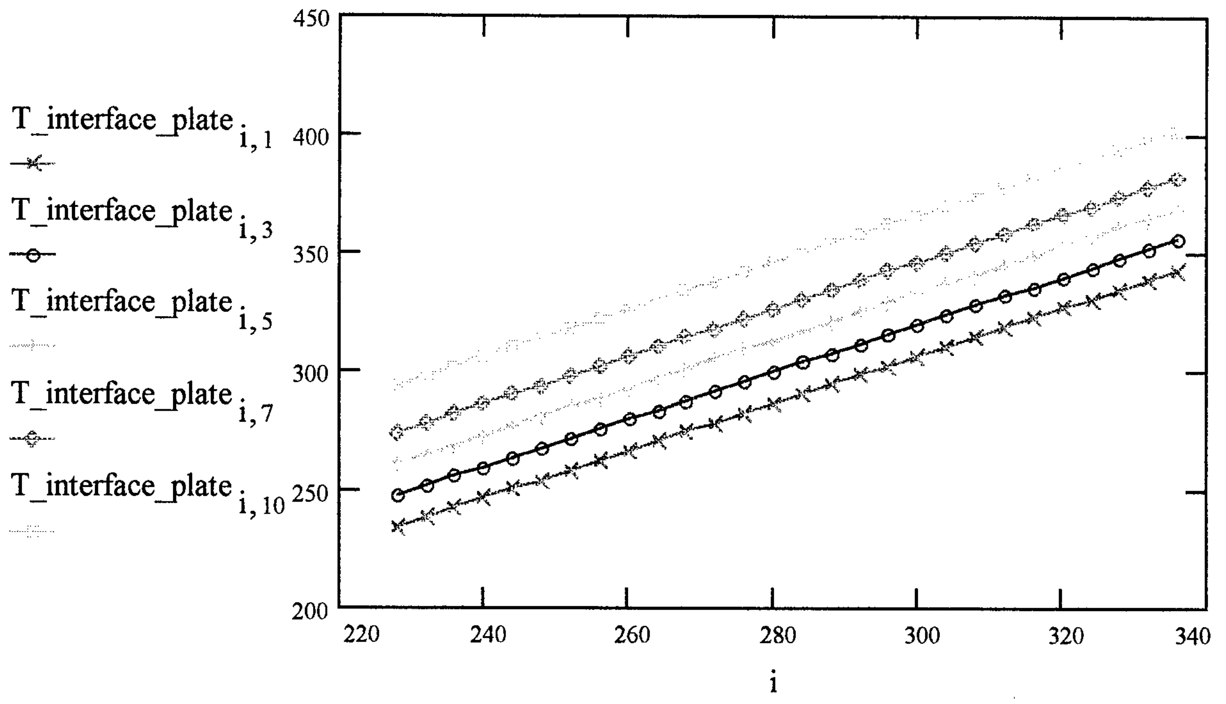
$L\_woven\_insulation_j = j \cdot cm$ $\quad A\_woven\_insulation = (43.63\,cm)^2 \cdot \pi$

$$T\_woven\_insulation_{i,j} := \left(\frac{Q\_dot \cdot L\_woven\_insulation_j}{k\_woven\_insulation \cdot A\_woven\_insulation} \right) + T\_internal\_sat_i$$



$L\_aluminum = 1 \cdot cm$ $\quad A\_aluminum = A\_woven\_insulation$

$$T\_interface\_plate_{i,j} := \left(\frac{Q\_dot \cdot L\_aluminum}{k\_aluminum \cdot A\_aluminum} \right) + T\_woven\_insulation_{i,j}$$


T_interface_plate i, 1
T_interface_plate i, 3
T_interface_plate i, 5
T_interface_plate i, 7
T_interface_plate i, 10
450
400
350
300
250
200
220
240
260
280
300
320
340
i

Bibliography

1. The Aerospace Corporation. Consolidated Mission Architecture. The Aerospace Corporation, El Segundo, California, Oct-93.

2. Aerospace Division Olin Defense System Group. Hydrazine Handbook. Redmond, WA: Rocket Research Company, 1995.

3. Aerospace Systems Division, 1996: http://www.ball.com/corporate/haerobas.html.

4. American Institute of Aeronautics and Astronautics, Inc., Phillips Laboratory Homepage, 1996: http://www.plk.af.mil.

5. Arizona State Univ. Statement of Work for the MightySat Phase II Program (Revision 4). Phillips Laboratory, Kirtland AFB, New Mexico, 31-Mar-96.

6. Asker, James R. "Ball Drops Pegasus Plan, Selects Lockheed Launcher," Aviation Week & Space Technology, : 58 (December 6, 1993).

7. Asker, James R. " Quick Response Key to Next U.S. Launcher," Aviation Week & Space Technology, : 44-46 (September 28, 1992).

8. Asker, James R. " Speedy Fix Eyed For Pegasus XL," Aviation Week & Space Technology, : 21-22 (July 3, 1995).

9. Athey, Thomas H. Systematic Systems Approach: An integrated Method for Solving System Problems. New Jersey: Prentice-Hall, 1982.

10. Bate, Roger R., Mueller, Donald D., and White, Jerry E. Fundamentals of Astrodynamics. New York: Dover Publications, 1971.

11. Bearden, David, and others. "Cost Modeling," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

12. Beer, Ferdinand P. and Johnston, E. Russell. Aircraft Structures. USA: McGraw-Hill Book Company, 1982.

13. Berget, Richard T., and Turner, Tim. "Command and Data Handling," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

14. Bible, Steven R. "The World Wide Web Amateur Satellite Ground Station," Proceedings of the AMSAT-NA 13th Space Symposium. Newington, CT: American Radio Relay League, 1995.

15. Boas, Mary L. Satellite Orbit Analysis Program (SOAP) Manual. The Aerospace Corporation, El Segundo, California, Apr-96.

16. Boden, Daryl G. "Introduction to Astrodynamics," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

17. Bonometti,R.J., and others. "DARPA Initiatives in Small-satellite Technology", SPIE Proceedings: Small Satellite Technology and Applications, Vol.1495: (1991).

18. Broadsky, R. F. "Defining and Sizing Space Payloads," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

19. Bruno, Pattan. Satellite Systems Principles and Technologies. NY: Van Nostrand Reinhold, 1990.

20. Chaggaris, Elias. Kearfott Capabilities in Space. Wayne, NJ: Kearfott Guidance and Navigation Corporation, 1996.

21. Chaggaris, Elias. Space Qualified Kearfott Inertial Reference Unit. Wayne, NJ: Kearfott Guidance and Navigation Corporation, 1996.

22. Chiang, J. J. and Kim, D. J. Mechanics of Materials. Boston, Massachusetts: PWS Engineering, 1984.

23. Chow, Brian G. An Evolutionary Approach to Space Launch Commercialization. Santa Monica, CA: RAND, 1993.

24. Clemen, Robert T. Making Hard Decisions: An Introduction To Decision Analysis. Belmont, CA: Duxbury Press, 1996.

25. The Clementine Team. Clementine II Concept Design Review. Naval Research Laboratory, CA. 14-Mar-96.

26. Davies, Richard S. "Communication Architecture," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

27. Dawdy, A.B. and Holker, D. L. An Approach to Rapid Payload Integration: The Spacecraft-To-Payload Interface Guideline (SPIG) Version 1. The Aerospace Corporation, El Segundo, California, Apr-96.

28. Day, Gregory R. and Lewis, Kelly D. "Strength Analysis". Spacecraft Structures and Mechanisms. Ed. Sarafin, Thomas P..Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

29. Department of the Air Force. FLTSATCOM Orbital Operations Handbook. The U.S. Air Force, 1977.

30. Dezelan, R. W. Cost Savings Recommendations for Warfighters Integrated Next Generation Space Systems (WINGSS). The Aerospace Corporation, El Segundo, California, Mar-96.

31. Dezelan, R. W. Distributed Satellite Formation Alternatives. The Aerospace Corporation, El Segundo, California, Mar-96.

32. Dornheim, Michael A. " Lockheed Must Prove LLV Reliability by June," Aviation Week & Space Technology, : 18-20 (August 21 1995).

33. Dornheim, Michael A. " Lockheed Completes LLV Mockup Stacking," Aviation Week & Space Technology, : 69 (August 1, 1994).

34. Dornheim, Michael A. " Range Video Captures LLV-1 Destruction," Aviation Week & Space Technology, : 53 (September 4, 1995).

35. Dornheim, Michael A. " Vectoring, IMU Destroyed LLV-1," Aviation Week & Space Technology, : 96-97 (December 18/25, 1995).

36. Doukas, Peter G., McCandless, James R., William R.Sarafin, and Thomas P. "Structures and Mechanisms," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

37. Doukas, Peter G., McCandless, James R.Britton, William R.Sarafin, and Thomas P. "Spacecraft Configuration and Structural Design". Fundamentals of Space Systems. Ed. Pisacane, Vincent L. and Moore, Robert C. New York: Oxford University Press, 1994.

38. Doyle, Christine. Space Inertial Reference Unit. Goleta, CA: Litton Guidance and Control Systems, 1996.

39. Dueber, Ross E. and McKnight, Darren S. Chemical Principles Applied To Spacecraft Operations. Malabar, Florida: Krieger Publishing Comp., 1993.

40. Eshbach, Ovid W. Mathematical Methods in Physical Sciences. NY: John Wiley & Sons, 1983.

41. Eterno, John S. "Attitude Determination and Control," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

42. Everett, James. Systems Engineer for Small Explorer program, NASA Goddard Space Flight Center Telephone interview. 3 October 1996.

43. Fleeter, Rick, and Warner, Richard. "Design of Low-Cost Spacecraft," Space Mission Analysis and Design. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1992.

44. Fleeter, Rick. "Reducing Spacecraft Cost," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

45. Ford, John. "Communications," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

46. Gere, James M. and Timoshenko, Stephen P. Mechanics of Materials. USA: McGraw-Hill Book Company, 1981.

47. Gorney, D.J., Blake J.B., Koons, H.C., Schultz, M., and Vampola, A.L. "The Space Environment and Survivability," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

48. Hagan, Joel. "Space Technology". Electronic mail. 16:40:14 EST, 11 October1996.

49. Hall III, Arthur D. "Three Dimensional Morphology of System Engineering", IEEE Transactions on System Science and Cybernetics, SSC-5, No.2: 156-160 (Apr-69).

50. Harvey, E.L. "Low-cost Spacecraft Buses for Remote Sensing Applications", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

51. Hecht, Herbert. "Reliability During Space Mission Concept Exploration," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

52. Hecht, Herbert. "Reliability Considerations," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

53. Hecht, H. and Hecht, M. Reliability Prediction for Spacecraft. Rome Air Development Center, NY. December 1985 (RADC-TR-85-229)

54. Higbee, Terry A. "TECHSTARS: Small, Smart Space Systems," Small-Satellite Technology and Applications. 103-114. Washington: SPIE-The International Society for Optical Engineering, (4-5 April) 1991.

55. High Performance Gyro Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

56. High Torque Momentum Reaction Wheel Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

57. Hively, Will. "Reinventing the Wheel", Discover, . Aug 96, pp. 58-68.

58. Ithaco Space Systems, 1996: http://www.newspace.com/Industry/Ithaco/home.html.

59. Jackson, Jon D. LCOL, USAF (SMC/TE) Telephone interview. 24 October 1996.

60. Jeffrey, W. "Small Satellites (MSTI-3) for Remote Sensing: Pushing the Limits of Sensor and Bus Technology", SPIE Proceedings: Platforms and Space Systems, Vol. 2317. (1995).

61. Jet Propulsion Laboratory. Flight Hardware Survey (Revision # 1). Pasadena, CA: Jet Propulsion Laboratory, 1993.

62. Kilston, S., and others. "Multi-mission Sensor for the RESERVES Small-satellite Program", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

63. Kitts, C.A. "Initial Developments in the Stanford Squirt Program", SPIE Proceedings: Platforms and Space Systems, Vol. 2317. 1995.

64. Kitts, C.A. "Integrated Control of Satellite Payloads", SPIE Proceedings: Platforms and Space Systems, Vol. 2317. 1995.

65. Kelements, H.D. Air Force Satellite Control Network Space and Ground Interface, Aerospace report No.TOR-0059(6110-01)-3. Contract No.F04701-88-C-0089. Satellite Control Network Directorate Satellite Data Division, March 1992.

66. Kolcum, Edward H. " NASA, Pentagon Chart Ambitious Unmanned Launch Vehicle Program," Aviation Week & Space Technology, : 131-133 (March 16, 1992).

67. Kostishack, D.F. "Small-satellite Sensors for Multispectral Space Surveillance", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

68. Kramer, Stuart. "Application of Concept Mapping To Systems Engineering," IEEE International Conference on Systems, Man, and Cybernetics Conference Proceedings. 652-654. New York: IEEE Press, 1990.

69. Kramer, Stuart. Vector Dynamics. Dayton, OH: Private Printing, 1996.

70. Landsat. "Landsat: What other spacecraft aspire to be... LANDSAT spacecraft and network". http://l7oe01.atsc.allied.com/mainpage.html.

71. Larson,Wiley J. and Wertz, James R. "The Space Mission Analysis and Design Process," Space Mission Analysis and Design. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1992.

72. Larson,Wiley J. and Wertz, James R. Reducing Space Mission Costs. Torrance, California: Microcosm Press, 1996.

73. Lenorovitz, Jeffrey M. " January Pegasus Launch Key to Mission Rate Build-Up," Aviation Week & Space Technology, : 69 (December 7, 1992).

74. Leritz, John E. and Palmer, Donald L. "Mechanics of Materials". Spacecraft Structures and Mechanisms. Ed. Sarafin, Thomas P..Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

75. Lockheed Martin. Lockheed Martin Launch Vehicle (LMLV) Capabilities Summary, 1996.

76. London III, John R. Leo On the Cheap. Air University press, Maxwell AF Base, Alabama, Oct. 1994.

77. Low Cost Gyro Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

78. Low Cost Momentum Wheel Assembly. Guidance and Control for Space Applications. Teterboro, NJ: Allied Signal Aerospace, 1996.

79. Maquire, Dick. Handbook of Engineering Fundamentals. USA: John Wiley & Sons, 1975.

80. McCall, G., and others. New World Vistas: Air and Space Power for the 21st Century, Summary Volume. Air Force Scientific Advisory Board, Albuquerque, 1995.

81. McDermott, Joseph K. "Power," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

82. McMordie, Robert K. "Thermal," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

83. McMordie, Robert K. "Configuring a Spacecraft," Spacecraft Structures and Mechanisms. Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

84. Mensa, Dean L. High Resolution RADAR Imaging. Norwood, MA: Artech House, 1981.

85. Meyers, Marc A. and Chawla, Krishan K. "Conceptual Design of Structures," Spacecraft Structures and Mechanisms. Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

86. MightySat Program Office. Statement of Work Appendix A Technical Requirements Document (TRD) for the MightySat Phase II Program (Revision 3). Phillips Laboratory, Kirtland AFB, New Mexico. 19-Apr-96.

87. Miniature Inertial Measurement Unit. Honeywell Satellite Systems. Glendale, AZ: Honeywell Space Systems Inc., 1996.

88. "Model Error Led to Pegasus Failure," Aviation Week & Space Technology, : 28 (August 1 1994).

89. Morgan, Walter L.and Gordon, Gary D. Principles of Communication Satellites. New York: John Wiley and Sons, Inc.1993.

90. Mosard, Gil R. "A Generalized Framework and Methodology for Systems Analysis," IEEE Transaction on Engineering Management, EM-29: 81-87 (Aug 1982).

91. Mosier, M.R., and others. "Pegasus Air-launched Space Booster Payload Interfaces and Processing Procedures for Small Optical Payloads," SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

92. Muolo, Michael J. Space Handbook: An Analyst's Guide. Air University press, Maxwell AFB, AL: Air University Press, Dec 1993.

93. Muolo, Michael J. Space Handbook: A Warfighter's Guide to Space (Vol.1). Maxwell AFB, AL: Air University Press, Dec 1993.

94. Myers, Raymond H. and Montgomery, Douglas C. Response Surface Methodology. New York: Jon Wiley & Sons, Inc., 1995.

95. The NASA Home Page, 1996: http:www.nasa.gov.

96. The NASA, ASUSat Homepage, 1996: enws229.eas.asu.edu/asusat/asusat.shtml

97. The NASA/JPL Imaging Radar Home Page. "The NASA/JPL Imaging Radar Home Page". http://southport.jpl.nasa.gov/desc/imagingradarv3.html.

98. The NASA/JPL SAR. "Miscellaneous issues on Synthetic Aperture RADAR (SAR)". http://southport.jpl.nasa.gov/.

99. NASA Launch Task Group. Future of the US Space Launch Capability: A Task Group Report. NASA, Washington, DC, Nov-92.

100.NASA Scientific and Technical Information Office. Technical Memorandum 4679: Space-borne Synthetic Aperture RADAR: Current Status and Future Directions. NASA, Washington, DC, 1995.

101.The NASA Small Explorer Program, 1996: www.laafb.af.mil/SMC.

102.Negron, David and Chomas, Arthur. "Mission Operations," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

103.Neter, John and others. Applied Linear Statistical Models. USA: Richard D. Edwin, Inc.,1990.

104.Nguyen, P., and others. Unmanned Space Vehicle Cost Model, 7th ed. Space And Missile Systems Center, Los Angeles, 1994.

105.Nicastri, Edward (Col. USAF Retiered). Former Director of DARPA's ATSSB program. Telephone interview. 5 June 1996.

106.Nicastri, Edward (Col. USAF Retiered). Former Director of DARPA's ATSSB program. Telephone interview. 30 July 1996.

107.Nicastri, Edward (Col. USAF Retiered). "Tactical Maneuvering, Payload Sizing, and Reliability issues". Electronic mail. 18:43:22 EST, 22 October1996.

108.Nozetto, S. and others. "Clementine Engineering and Technology Workshop," Proceedings of The Clementine Program's Engineering and Technology Workshop. Washington DC: Department Of Defense Ballistic Missile Defense Organization, 26 July 1994.

109.O'Neil, J., and others. "Eagle-class Small Satellite for LEO Applications", SPIE Proceedings: Small Satellite Technology and Applications, Vol. 1495: (1991).

110.Orbital Sciences Corporation, 1996: http://www.orbital.com.

111.Orbital Science Corporation. Commercial Pegasus Payload User's Guide. Office of Launch Vehicle, Business Development, Pegasus Launch Vehicle Systems, Dulles, Virginia. 1-Oct-93.

112.Orbital Science Corporation. Commercial Taurus launch Vehicle Payload User's Guide. Office of Launch Vehicle, Business Development, Taurus Launch Vehicle Systems, Dulles, Virginia.18-Apr-96.

113." Pegasus Carrier Aircraft Completes Flight Tests," Aviation Week & Space Technology, : 57-58 (December 6, 1993).

114.Pegasus Marketing Video. Prod. Orbital Science Corporation. Dulles, Virginia. 1995.

115.Peery, David J. and Azar, J. J. Mechanical Metallurgy. Englewood Cliffs, NJ: Prentice-Hall, Inc., 1984.

116.Pohl, Ed. Class handout, SENG 520.School of Engineering, Air Force Institute of Technology, WPAFB OH. 6-Oct-95.

117. Presley, S.P., Agardy, F.J., and Berrier, D.J. An Appoach To Rapid Payload Integration: The Spacecraft-To-Payload Interface Guideline (SPIG). The Aerospace Corporation, El Segundo, CA. 8-Apr-96.

118. Rademacher, Joel. "NASA Small Explorer (SMEX) Home page: NASA Small Explorer Program," 1996: sunland.gsfc.nasa.gov/smex/smexhomepage.html.

119. Rademacher, Joel. "Phillips Laboratory," 1996: enws229.eas.asu.edu/joel/joel.shtml

120. Ramakumar, R. Engineering Reliability - Fundamentals & Applications. Englewood Cliffs, CA: Prentice Hall, 1993.

121. Rash, James L. Third International Symposium on Space Missions and Ground Data Systems Part I/II. Greenbelt, Maryland: NASA, Nov 15-18, 1994.

122. Reaction Wheel and Momentum Wheel Assemblies. Space Systems - Momentum Control. Glendale, AZ: Honeywell Space Systems Inc., 1996.

123. Rees, W.G. Physical Principles of Remote Sensing. Cambridge: Cambridge Univ. Press, 1990.

124. Reeves, Emery "Spacecraft Design and Sizing," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

125. Reeves, Emery "Spacecraft Manufacture and Test," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

126. Reichhardt, Tony. " Launch Woes Ground NASA Science Spacecraft," Naure, 45.717 (August 31, 1995).

127. Reinert, Richard P. and Wertz James R. "Mission Characterization," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R.. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

128. Reynolds, William C. and Perkins, Henry C. Tactical Support Satellite/Standard Payload Interface (TSS/SPI) Study: Standard Interface Definitions and Cost Comparisons. The Aerospace Corporation, El Segundo, California, Sep-94.

129. Sackheim, Robert L. "Space Propulsion System," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

130. Sarafin, Thomas P., Heymans, Robert J.Wendt, Robert G. Jr., and Sabin, Robert V. "Thermal Effects," Spacecraft Structures and Mechanisms. Ed. Sarafin, Thomas P..Torrance- California / Boston: Microcosm, Inc. / Kluwer Academic Press (jointly), 1995.

131. Sarafin, Thomas P. AIAA Aerospace Design Engineers Guide. American Institute of Aeronautics and Astronautics, Inc., Washington DC, 1993.

132. Scott, Pace. US Access to Space Launch Vehicle Choices for 1990-2010. RAND Santa Monica, CA, Mar-90.

133. Sellers, Jerry, and Milton, Ed. "Technology for Reduced Cost Missions," Reducing Space Mission Cost. Ed. Wertz, James, and Larson, Wiley. Torrance, CA: Microcosm, Inc., 1996.

134. Servo Space Systems, 1996: http://www.servo.com/spasys.html.

135. Shuster, Malcolm D. Fundamentals of Space Systems. New York : Oxford University Press, 1994.

136.Skullney, W. E. "Spacecraft Thermal Control". Fundamentals of Space Systems. Ed. Pisacane, Vincent L. and Moore, Robert C. New York: Oxford University Press, 1994.

137.Smith, Bruce A. " Orbital L-1011 Ready for First Launch," Aviation Week & Space Technology, : 30 (July 4, 1994).

138.Smith, Bruce A. " OSC Seeks Cause of Pegasus XL Failure," Aviation Week & Space Technology, : (1994).

139.Smith, Bruce A. " Pegasus XL Success Opens Door to Fast-Paced Launch Schedule," Aviation Week & Space Technology, : 26-27 (March 18, 1996).

140.Space & Missile Systems Center, Orbital Sciences Corporation's (OSC's) Home Page, 1996: www.orbital.com

141.Space Systems Division. Pegasus Design Safety Requirements Document. Orbital Science Corporation, Fairfax, Virginia, 27-May-92.

142.Stanford SSDL Home Page. "Stanford SSDL Home Page. Space System Design at Stanford University. http://aa.stanford.edu/~ssdl/.

143.Stodden, David Y. and Galasso, Gina D. Engineering Theromodynamics. USA: McGraw-Hill Book Company, 1977.

144.Sutton, George P. Rocket Propulsion Elements. New York: Jonh Wiley and Sons, 1992.

145.Taurus Marketing Video. Prod. Orbital Science Corporation. Dulles, Virginia. 1996.

146.Telemetry Components and Systems, Command, Control and Communications Catalog. Newtown, PA. AYDIN VECTOR, 1996.

147.The Aerospace Corporation. Communications Satellites 1958-1992. El Segundo, California: The Aerospace Corporation, 1991.

148.U.S. Air Force Home Page, 1996: http://www.af.mil.

149.U.S. Congress, Office of Tecnology Assesment. Affordabble Spacecraft Design and Launch Alternatives. Washington, DC: US printing Office. (OTA-BP-ISC-60)

150.Vesely, David. "Transcript : DOD News Briefing : Major General David Vesely, 14th Air Force Commander. Miscellaneous in-theater space operations concepts". http://www.laafb.af.mil/SMC/CZ/homepage/news/dodnew13.htm.

151.Velocci, Anthony L. " Orbital Says Human Error Caused Pegasus XL Loss," Aviation Week & Space Technology, : 60 (September 4, 1995).

152.Walker, Kevin J. Systems Analysis of GPS Electrical Power System Redesign. MS thesis, AFIT/ENY/AFIT/GSO/ENY/95D-03. School of Engineering, Air Force Institute of Technology, WPAFB OH, Dec-95.

153.Warner, Richard. Founder and Chief Technical Officer of AeroAstro Corp. Personal interview. 18 July 1996.

154.Wertz, James R. "Mission Evaluation," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R.. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

155.Wertz, James R. "Space Mission Geometry," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

156.Wertz, James R. "Orbit and Constellation Design," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

157.Wiesel, William E. Spaceflight Dynamics. New York: McGraw-Hill, 1989.

158. Williams, Michael S. "Requirements Definition," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

159. Wingate, Clarence. SatAC Materials Database (matter.dat). Phillips Laboratory, «Publisher_location», Mar-96.

160. Wong, Robert "Cost Modeling," Space Mission Analysis and Design. Ed. Larson, Wiley J. and Wertz, James R. Torrance CA / Boston: Microcosm Inc. and Kluwer Academic Publishers (jointly), 1992.

161. Worden, Col. Simon P.,COL, USAF (50SW/CC)."Space: Grabbing the Solar System (and Dominating the Planet)," AF 2025 Submission. Falcon AFB, CO: 50th Space Wing, 1996.

Vita

Capt Gerald (Gerry) F. Ashby [redacted]

After graduating from Riverside High School in June 1983, he entered the enlisted ranks of the U.S. Air Force. A year later he entered the U.S.A.F Academy Preparatory School in Colorado Springs, Colorado. Upon graduating with honors, he entered the U.S. Air Force Academy and joined the class of 1989. Four years later on 31 May 1989, graduating in the top 15% of his class, he received his degree in Engineering Sciences of Aerospace Structures and his regular commission in U.S. Air Force.

Upon graduation, he left for pilot training at Williams AFB, Arizona and after unsuccessfully completing the program, he entered Undergraduate Space Training at Lowry AFB in August 1990. His next assignment was to Falcon AFB as a Launch Analyst for the Navy's UHF F/O communication satellite program. After leaving Falcon AFB in December 1993, he transferred to Detachment 1, AFSPC at Los Angeles AFB, California, where he worked as the main Air Force Space Command spokesman within the Milsatcom Joint Program Office. Then in May 1995, he entered the School of Engineering, Air Force Institute of Technology.

[redacted]

Captain Darren J. Buck [redacted]. He graduated from Holy Cross High School in Dover, Delaware May 1986 and began the Freshman Year Studies at the University of Notre Dame in August 1986. He graduated from Notre Dame with a Bachelor of Science degree in Mathematics in May 1990 . He received his commission on 19 May 1990 after having completed the Notre Dame Air Force Reserve Officer Training Corps (NDAFROTC).

After completing Undergraduate Space Training at Lowry AFB, CO in August 1991, his first assignment was as a Satellite Operations Crew Commander at the 3rd Satellite Control Squadron (3 SCS -- later the 3rd Space Operations Squadron (3 SOPS) at Falcon AFB, CO. His two subsequent assignments, also at Falcon, were as a Simulation Software Development/QAE Officer and Satellite Systems Instructor for the MILSTAR Satellite Program at the 50th Crew Training Squadron (50 CTS), and as Group Operations Training Officer for MILSTAR at the 50th Operations Support Squadron (50 OSS). In May 1995, he entered the Graduate Space Operations Program in the School of Engineering at the Air Force Institute of Technology.

[redacted]3

Lieutenant Robert Carneal IV [redacted]

He graduated from two high schools in 1988 located in Virginia: Thomas Jefferson School for Science and Technology, and Fairfax High School. In the Fall of 1988 he entered the Prescott, AZ campus of Embry-Riddle Aeronautical University. He graduated with honors with a bachelor's science degree in electrical engineering, with a math minor, in December 1992. Upon graduation he was commissioned second lieutenant in the USAF.

His first assignment was the 5th Space Launch Squadron at Cape Canaveral Air Station, where he was involved in launching Titan IV rockets. In May 1995 he entered the Air Force Institute of Technology at Wright-Patterson as a masters candidate in space operations. His follow on assignment is with the Space Warfare Center at Falcon AFB.

[redacted]

Lieutenant Tansel Cokuysal [redacted] He graduated from Maltepe Military High School in 1987 and entered undergraduate studies at the Turkish Air Force Academy in Istanbul, Turkey. He graduated with a Military of Science degree in Aerospace Engineering August 1991. Upon graduation, he was commissioned a Second Lieutenant in Turkish Air Force.

His first assignment was at 2[nd] Main Jet Base, Izmir Turkey for Basic Jet Pilot Training. His second assignment was to 3[rd] Main Jet Base, Konya Turkey for F-5A/B Combat Readiness Training. His third assignment was to 4[th] Main Jet Base, Ankara Turkey for F-16 Basic and Combat Readiness Training. His subsequent assignment was to 9[th] Main Jet Base, Balikesir Turkey as a Wing-man in 191[st] (Cobra) Squadron. In May 1995, he entered the School of Engineering , Air Force Institute of Technology.

[redacted]

Lieutenant Ahmet Tuna Donmez [redacted].

He graduated from Maltepe Military High School, Izmir in 1987 and entered undergraduate studies at the Turkish Air Force Academy in Istanbul, Turkey. He graduated with a Military of Science degree in Aerospace Engineering August 1991. Upon graduation, he was commissioned as a Second Lieutenant in Turkish Air Force.

His first assignment was at 2nd Main Jet Base, Izmir Turkey for Basic Jet Pilot Training. His second assignment was to 3rd Main Jet Base, Konya Turkey for F-5A/B Combat Readiness Training. His third assignment was to 1st Main Jet Base, Eskisehir Turkey for F-4E and RF-4E Basic and Combat Readiness Training. His subsequent assignment was to 1st Main Jet Base, Eskisehir Turkey as a Wing-man in 113th Tactical Reconnaissance Squadron. In May 1995, he entered the School of Engineering , Air Force Institute of Technology to have his Maters of Science degree in System Engineering.

[redacted]

Capt James A. From [redacted] He graduated from Frank W. Cox High School in 1981 and entered undergraduate studies at the University of Virginia in Charlottesville, Virginia in 1982. He graduated with a Bachelor of Science degree in Aerospace Engineering in May 1987. He received his commission on 16 May 1987 and was a Distinguished Graduate from Reserve Officer Training Corps.

His first assignment was at Lowry AFB for Undergraduate Space Training. His second assignment was to Peterson AFB as a Defense Satellite Communications System phase III (DSCS III) Commander and Planner/Analyst Instructor. His subsequent assignment was to Buckley ANGB as a Satellite Operations Crew Commander and later Chief of Standardization and Evaluation for the Defense Support Program. While at Buckley ANGB, he earned a Masters of Arts degree in Space Systems Management from Webster University. In May 1995, he entered the School of Engineering, Air Force Institute of Technology. His follow on assignment is to the United States Space Command Headquarters at Peterson AFB.

[redacted]

Capt Todd C. Krueger [redacted]. He graduated from Sprague High School in 1987 and entered undergraduate studies at the University of Southern California in Los Angeles, California. He graduated with a Bachelor of Science degree in Aerospace Engineering in May 1991. He received his commission on 9 May 1991, having completed the Air Force Reserve Officer Training Corps (AFROTC) program.

His first assignment was at Falcon AFB as a satellite analyst/instructor. In May 1995, he entered the School of Engineering, Air Force Institute of Technology.

[redacted]

First Lieutenant Brian I. Robinson [redacted].
He graduated from Whitney M. Young High School in 1987, and earned a Bachelor of Science degree in Aerospace Engineering with honors, and an Air Force commission (Distinguished Graduate) from Tuskegee University in May 1993.

His only assignment prior to AFIT was as an engineer for the Titan System Program Office (now the Launch Programs System Program Office), Los Angeles Air Force Base, CA, from September 1993 to May 1995. His duties included managing the ongoing development, production, acquisition, and delivery of the Titan IV's Solid Rocket Motors (SRM), and was present at Cape Canaveral AS for five Titan IV launches, serving as the Titan SPO's system expert on the Titan SRMs.

Lt Robinson is married to Lt Coleen Y. Robinson, USAF.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
November 1996

3. REPORT TYPE AND DATES COVERED
Master's Thesis

4. TITLE AND SUBTITLE
The Preliminary Design of a Standardized Spacecraft Bus for Small Tactical Satellites (Volume 2)

5. FUNDING NUMBERS

6. AUTHOR(S)
Gerald F. Ashby, Capt, USAF, et al.

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Air Force Institute of Technology, WPAFB OH 45433-6583

8. PERFORMING ORGANIZATION REPORT NUMBER
AFIT/GSE/GSO/ENY/96D-1

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
Lt Col James Rooney
Phillips Laboratory/WSM
3550 Aberdeen Ave SE
KAFB, NM 87117-5776

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Approved for public release; distribution unlimited

12b. DISTRIBUTION CODE
A

13. ABSTRACT *(Maximum 200 words)*

Current satellite design philosophies concentrate on optimizing and tailoring a particular satellite bus to a specific payload or mission. Today's satellites take a long time to build, checkout, and launch. An alternate approach shifts the design paradigm to one that focuses on access to space, enabling tactical deployment on demand and the capability to put current payload technology into orbit, versus several years by today's standards, by which time the technology is already obsolete. This design study applied systems engineering methods to create a satellite bus architecture that can accommodate a range of remote sensing mission modules. System-level and subsystem-level tradeoffs provided standard components and satellite structures, and an iterative design approach provided candidate designs constructed with those components. A cost and reliability trade study provided initial estimates for satellite performance. Modeling and analysis based upon the Sponsor's objectives converged the designs to an optimum solution. Major products of this study include not only a preliminary satellite design to meet the sponsor's needs, but also a software modeling and analysis tool for satellite design, integration, and test. Finally, the report provides an initial implementation scheme and concept for operations for the tactical support of this satellite system.

14. SUBJECT TERMS
Satellite Bus, Spacecraft Design, Systems Engineering, Modular Satellite, Satellite Subsystems, Remote Sensing

15. NUMBER OF PAGES
451

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	UL

GENERAL INSTRUCTIONS FOR COMPLETING SF 298

The Report Documentation Page (RDP) is used in announcing and cataloging reports. It is important that this information be consistent with the rest of the report, particularly the cover and title page. Instructions for filling in each block of the form follow. It is important to *stay within the lines* to meet *optical scanning requirements.*

Block 1. Agency Use Only *(Leave blank).*

Block 2. Report Date. Full publication date including day, month, and year, if available (e.g. 1 Jan 88). Must cite at least the year.

Block 3. Type of Report and Dates Covered. State whether report is interim, final, etc. If applicable, enter inclusive report dates (e.g. 10 Jun 87 - 30 Jun 88).

Block 4. Title and Subtitle. A title is taken from the part of the report that provides the most meaningful and complete information. When a report is prepared in more than one volume, repeat the primary title, add volume number, and include subtitle for the specific volume. On classified documents enter the title classification in parentheses.

Block 5. Funding Numbers. To include contract and grant numbers; may include program element number(s), project number(s), task number(s), and work unit number(s). Use the following labels:

C	- Contract	PR	- Project
G	- Grant	TA	- Task
PE	- Program Element	WU	- Work Unit Accession No.

Block 6. Author(s). Name(s) of person(s) responsible for writing the report, performing the research, or credited with the content of the report. If editor or compiler, this should follow the name(s).

Block 7. Performing Organization Name(s) and Address(es). Self-explanatory.

Block 8. Performing Organization Report Number. Enter the unique alphanumeric report number(s) assigned by the organization performing the report.

Block 9. Sponsoring/Monitoring Agency Name(s) and Address(es). Self-explanatory.

Block 10. Sponsoring/Monitoring Agency Report Number. *(If known)*

Block 11. Supplementary Notes. Enter information not included elsewhere such as: Prepared in cooperation with...; Trans. of...; To be published in.... When a report is revised, include a statement whether the new report supersedes or supplements the older report.

Block 12a. Distribution/Availability Statement. Denotes public availability or limitations. Cite any availability to the public. Enter additional limitations or special markings in all capitals (e.g. NOFORN, REL, ITAR).

- DOD - See DoDD 5230.24, "Distribution Statements on Technical Documents."
- DOE - See authorities.
- NASA - See Handbook NHB 2200.2.
- NTIS - Leave blank.

Block 12b. Distribution Code.

- DOD - Leave blank.
- DOE - Enter DOE distribution categories from the Standard Distribution for Unclassified Scientific and Technical Reports.
- NASA - Leave blank.
- NTIS - Leave blank.

Block 13. Abstract. Include a brief *(Maximum 200 words)* factual summary of the most significant information contained in the report.

Block 14. Subject Terms. Keywords or phrases identifying major subjects in the report.

Block 15. Number of Pages. Enter the total number of pages.

Block 16. Price Code. Enter appropriate price code *(NTIS only).*

Blocks 17. - 19. Security Classifications. Self-explanatory. Enter U.S. Security Classification in accordance with U.S. Security Regulations (i.e., UNCLASSIFIED). If form contains classified information, stamp classification on the top and bottom of the page.

Block 20. Limitation of Abstract. This block must be completed to assign a limitation to the abstract. Enter either UL (unlimited) or SAR (same as report). An entry in this block is necessary if the abstract is to be limited. If blank, the abstract is assumed to be unlimited.